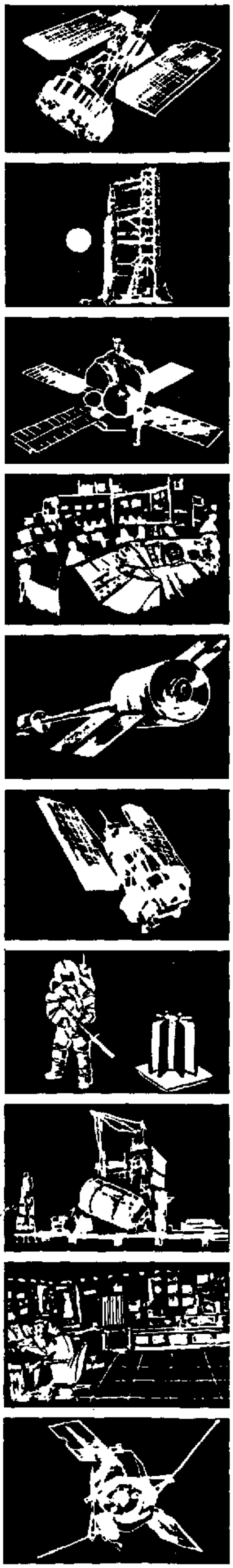
SPACE DIVISION



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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 1

ORBIT/LAUNCH VEHICLE TRADE-OFF STUDIES AND RECOMMENDATIONS

EOS EARTH OBSERVATORY SATELLITE

GENERAL ELECTRIC

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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 1

ORBIT/LAUNCH VEHICLE TRADE-OFF STUDIES AND RECOMMENDATIONS

Prepared for:
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Under
Contract No. NAS 5-20518

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SECTION 1.0

SUMMARY

This section presents a summary of the driving constraints/requirements on the EOS-A orbit/launch vehicle section, summarizes the propulsion system (Hydrazine) and launch vehicle (Delta 2910) selected for EOS-A and presents the rationale for selection of the recommended EOS-A orbit (418 nm). This summary is for the originally defined EOS-A mission including the Thematic Mapper and the High Resolution Pointable Imager. Finally, the impact of the revised mission model received on June 18/19 is discussed stating how the new mission model affects the previously defined propulsion system, launch vehicle and orbit.

The driving constraints for the orbit launch vehicle selection can be separated into three key areas that are:

- o mission constraints
- o launch system constraints
- o spacecraft weight constraints

The mission constraints that drive the orbit altitude selection can be summarized as:

o real time coverage of USA	
o sidelap width	5 to 15% (prefer 10% max)
o repeat cycle	15 to 18 days
o access time	2 to 4 days
o minimum offset pointing	$\leq 30^{o}$ (HRPI)

Five candidate orbits that meet the majority of these mission constraints have been identified. The altitudes of these candidate orbits range from 359 nm to 425 nm. The lower limit was selected as the practical minimum altitude due to drag effects while the upper altitude limit was set by the launch system performance.

The launch system constraints for the primary launch vehicles are shown in Table 1-1. Titan IIID NUS has been eliminated due to the high launch cost of between 25-44 M dollars.

The launch system costs (which include the on-board propulsion system) vary from $6.6 to $12.9M, while the allowable spacecraft weight varies from 2380 to 4520 lbs. depending on launch vehicle and mission altitude. The Titan shroud volume is approximately three times the volume of the Delta shroud.

Table 1-1. Launch System Cost/Weight/Volume Comparison

Launch Vehicle	Launch Vehicle Cost ('74 Dollars) M	Prop Syst. Cost M	Total Cost M	Shroud Volume FT^3	Allowable S/C * Weight (lbs) (Minus Propulsion)	
					350 nm	400 nm
Delta 2910	6.0	0.6	6.6	600	2520	2380
Delta 3910	8.0	0.6	8.6	600	3520	3340
Titan IIIB NUS	12.2	0.7	12.9	1670	4520	4340

* Allowable weights assume spacecraft returned to shuttle for retrieval.

The spacecraft weights for three alternate Delta spacecraft and two alternate Titan spacecraft containing varying levels of capability are shown in Table 1-2.

Table 1-2. EOS-A Spacecraft Weights (Delta & Titan S/C)

	Delta Spacecraft Wts (lbs)			Titan Spacecraft Wts (lbs)	
	Max Capability	Nominal Capability	Light Weight	Max Capability	Nominal Capability
Basic Spacecraft	1599	1472	1383	2123	1691
Mission Peculiar	576	343	160	576	343
Payload Instruments (TM, HRPI & DCS)	1205	937	672	1205	937
Total Spacecraft (minus propulsion)	3380	2752	2215	3904	2971

The Hydrazine propulsion system was selected due to its low cost for EOS-A in addition to its flexibility and low cost in meeting the requirements of the total mission model. The all hydrazine system proved lowest cost for Delta, Titan or Shuttle applications. The selection of the all hydrazine propulsion system is discussed further in Section 5.0 of this report and in Report #3.

The Delta 2910 was selected as the preferred launch vehicle for EOS-A since it is the lowest cost launch vehicle that can perform a meaningful EOS-A mission. The Delta 2910 can launch the light weight Delta spacecraft to 418 nm with sufficient spacecraft propulsion on-board to return the spacecraft to 330 nm for Shuttle retrieval. The 418 nm altitude is also directly Shuttle accessible (with a cost penalty) should recovery be required without use of the on-board propulsion system.

If retrieval of EOS-A is not required the allowable payload instrument weight can be increased from 670 lbs to 900 lbs while retaining a weight contingency of 140 lbs.

The mission orbit of 418 nm was selected as the best compromise between Shuttle compatibility, mission compatibility, ground system compatibility, launch system impacts, spacecraft impacts, payload instrument impacts and impacts of later missions and is summarized in Table 1-3. The orbit selection tradeoff matrix is presented in detail as Table 6-1 of this report and discussed in Section 6.0 along with the propulsion system and launch vehicle selection.

The revised mission model received on June 18/19 does not impact the selection of the on-board propulsion system since the selected hydrazine system provides ample flexibility to accommodate a wide range of missions. The revised mission model does however impact the launch vehicle and orbit selections in varying degrees. To establish these impacts the emphasis has been to evaluate the requirements for EOS-A (TM & MSS), EOS-A' (MSS) and the combined spacecraft (TM & 2 MSS). The weights for these three spacecraft are summarized in Table 1-4. It should be noted that a weight contingency of 10% has been included in the table and that the propulsion system weights have not been included since they are a function of orbit altitude and have been accounted for in the launch system capability curves.

Table 1-3. EOS-A Orbit Selection Summary

Evaluation Criteria	Candidate Orbits (nm)				
	359	386	399	418	425
Shuttle Compatibility	Good	Good	Good	Good	Fair
Mission Compatibility	Good	Fair	Fair	Good	Good
Ground Station Compatibility	Poor	Fair	Good	Good	Good
Launch System Impacts	Good	Good	Good	Fair	Fair
Spacecraft Impacts	Minor impacts over range of altitudes				
Payload Instrument Impacts					
Impacts of Later Missions					
				selected orbit	

Table 1-4. EOS Spacecraft Weights For Revised Mission Model

	Delta Spacecraft Weights (lbs)		
	EOS-A (TM & MSS)	EOS-A' (MSS)	Combined S/C (TM & 2MSS)
Basic Spacecraft	1217	1217	1217
Mission Peculiar	445	354	688
Payload Instruments	485	154	627
Weight Contingency	210	170	250
Total Spacecraft (minus propulsion)	2357	1895	2782

Figure 1-1 presents the launch vehicle capability (no spacecraft retrieval) as a function of mission altitude with the weights of the three alternate spacecraft superimposed. This figure illustrates that either the EOS-A or A' spacecraft can be launched with Delta 2910 to any mission altitude from 300 to 500 nm while the combined spacecraft with the Thematic Mapper and two multispectral scanners requires the Delta 3910 over the entire range of mission altitudes. Therefore, if there is no spacecraft retrieval required the choice of mission altitude (for the range between 300 and 500 nm) can be made without impacting the launch vehicle selection.

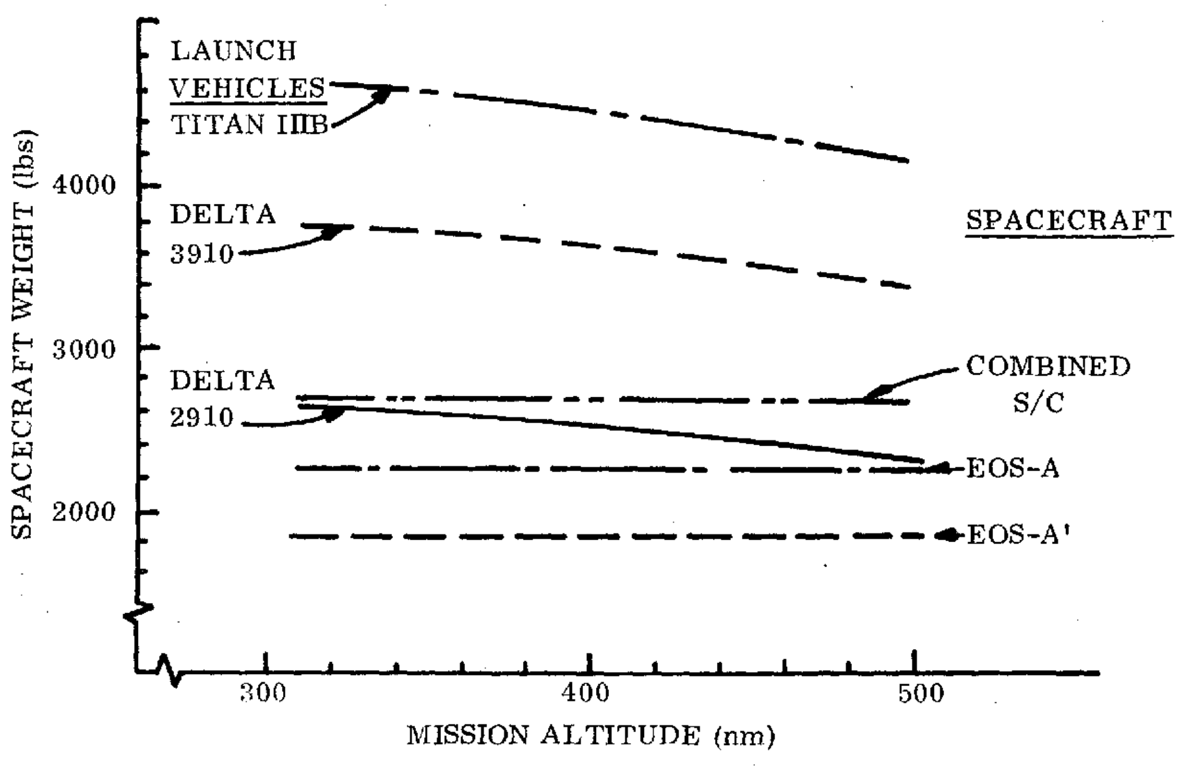


Figure 1-1. Launch System Performance and Spacecraft Weights for Revised Mission Model (No Retrieve of S/C)

The impact of spacecraft retrieval by Shuttle at 300 nm is illustrated in Figure 1-2. If spacecraft retrieval is required the following impacts are generated for EOS-A, EOS-A' and the combined spacecraft.

- o EOS-A' can be launched by Delta 2910 at mission altitudes over the entire range of 300 to 500 nm with Shuttle retrieval at 300 nm by use of an on-board propulsion system.
- o EOS-A can be launched by Delta 2910 to mission altitudes from 300 to 400 nm with Shuttle retrieval at 300 nm.
- o EOS-A requires a Delta 3910 launch if mission altitudes near 500 nm are required for a Shuttle retrievable spacecraft.
- o The combined spacecraft can be launched by Delta 3910 to mission altitudes over the entire range of 300 to 500 nm with Shuttle retrieval at 300 nm by use of an on-board propulsion system.

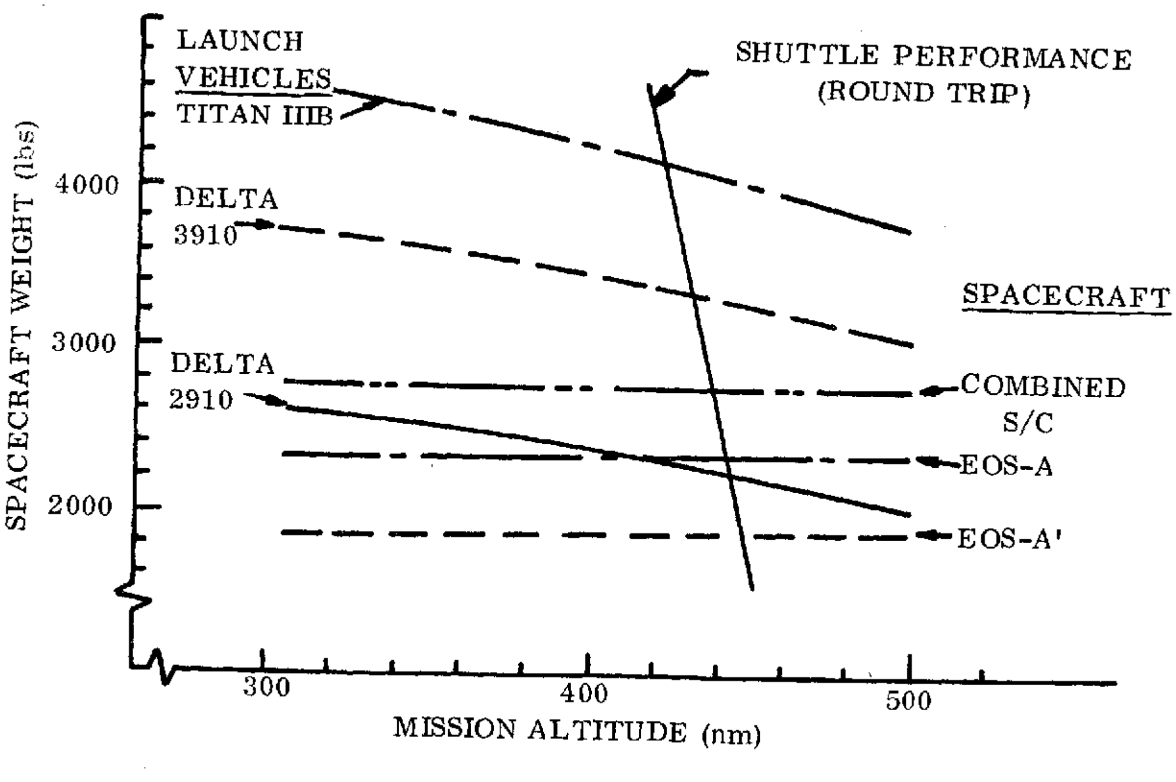


Figure 1-2. Launch System Performance and Spacecraft Weights for Revisied Mission Model (Return to Shuttle @ 300 nm)

The mission impacts of the new mission model are considerably more severe than the impacts on the spacecraft design and launch vehicle selection. The MSS is presently designed for an altitude of 496 nm. The instrument Instantaneous Field of View (IFOV) is a function of fiber optics size, telescope focal length and orbital altitude. If the orbital altitude is changed from 496 nm, the IFOV can be retained by either changing the telescope focal length (major redesign) or changing the fiber optics (smaller redesign). To minimize these impacts on the MSS it is advisable to select the mission orbit near 496 nm when MSS is included in the instrument payload. Missions are being considered where the altitude can be adjusted to provide varying repeat cycles, i.e., 18 and 9 days. The mission altitudes for these cases are 496 nm and 506 nm with these exact altitudes required to maintain the repeating orbit. The actual altitude of the spacecraft above the surface of the earth varies as a function of position in orbit and also varies as a function of long-term periodic effects. A considerable amount of analysis was performed on the ERTS program to understand this effect. Normal altitude variations about the nominal of approximately $\pm$ 10 nm can be expected. Therefore the altitude variation for the two orbits under consideration can be as great as 30 nm which may be difficult to accomodate without some instrument redesign. The impact of flying the MSS at orbits more compatible with the Thematic Mapper and HRPI are more complex and require further analysis to establish the cost effective approach.

SECTION 2.0

REQUIREMENTS/CONSTRAINTS/CRITERIA

2.1 INTRODUCTION

This section defines the requirements and constraints placed on the selection of the EOS-A orbits, launch vehicle and integral tug. The orbit selection is constrained by:

Mission Constraints

- Mission requirements for EOS-A which impact orbit parameter such as sidelap repeat cycle, access time, swath width and altitude range.
- Requirements for later missions such as EOS-B, C .... and missions such as Solar Maximum, Seasat, ERS, and SEOS which logically could be accommodated by a general purpose spacecraft such as EOS.

Launch Vehicle Constraints

- Physical shroud restrictions, environmental criteria, injection accuracy and type of guidance system and costs for a range of alternate launch vehicles.
- Compatibility with Space Shuttle for retrieval and/or servicing of early EOS launches plus launch, retrieval and/or servicing for later spacecraft when shuttle becomes operational.

Spacecraft Constraints

- Spacecraft weights for the basic spacecraft, mission peculiar equipment and payload instruments.
- Configuration constraints affecting the adaptability of the basic spacecraft to be packaged within a launch vehicle shroud or supporting an efficient mechanical interface to the alternate launch vehicle.
- Integral propulsion system designs that are required to augment the performance of some of the candidate launch vehicles to reach desired mission orbits and required in many orbits to return the spacecraft to preferred shuttle retrieval orbits.

Instrument Constraints

- Instrument constraints such as size, weight, power and orientation requirements which may eliminate some launch vehicles or orbits due to physical size incompatibilities with shrouds or weights limitation with launch vehicle performance capabilities.

This section also defines the evaluation criteria established to select the preferred orbit and launch vehicle for EOS-A. This criteria considers shuttle compatibility, mission compatibility, launch system impacts (pre-shuttle), spacecraft impacts and impacts of later missions.

2.2 MISSION CONSTRAINTS

In order to effectively design a general purpose spacecraft and perform trades to optimize program costs, it is necessary to understand the potential missions which the launch vehicle/ spacecraft system is expected to support. The RFP identified Solar Maximum, Seasat, ERS and SEOS as missions which potentially could use the EOS general purpose spacecraft. Information on these missions has been extracted from the reference documents supplied both with the RFP and since contract award. The extracted information is summarized in Table 2-1 and has been used as the point of departure for study of a general purpose spacecraft system. The table includes data on Seasat-B also since the planned launch date of Seasat-A may make it incompatible with the EOS development cycle.

The RFP also identifies candidate instruments such as the Synthetic Aperture Radar and Passive Multi-channel Microwave Radiometer for flight on EOS-B, C .... spacecraft. It was clear early in the study that the general purpose spacecraft can easily accommodate these instruments given that they fly individually on separate missions. It is also clear, however, that is is unlikely that these instruments will fly separately since additional payload capability is available. Even more importantly, there is much greater intrinsic value in the data from a complement of sensors over that provided from a single sensor. Hence realistic complements of sensors were postulated around the SAR and PMMR in keeping with the basic nature of the missions which the instruments are to support. These realistic payloads were then used as the point of departure for studies for follow-on EOS missions.

Briefly the missions considered are:

EOS-A. Land Resources Management, the primary mission from which the basic spacecraft performed requirements were drvied.

Follow-on EOS Missions

EOS-B. Tentatively planned to be an Oceanography/Meteorology mission with the Passive Multi-channel Microwave Radiometer as the primary payload sensor. Supporting sensors are consistent with long term ocean/met payload development goals.

TABLE 2-1. EOS MULTIPLE MISSION DATA

SYSTEM		EOS-A LAND RESOURCE MANAGEMENT	EOS-B OCEANOGRAPHY METEROLOGY	EOS-C ALL WEATHER OBSERVATORY	SHUTTLE RESUPPLY DEMONSTRATION TEST FLIGHT	SEOS	SOLAR MAX	SEASAT-A	SEASAT-B	5 BAND MSS
MISSION		DEVELOP ADVANCED INSTRUMENTS WHICH CAN PROVIDE MULTISPECTRAL IMAGERY OF THE LAND SURFACE AT SIGNIFICANTLY IMPROVED SPATIAL, SPECTRAL AND TEMPORAL RESOLUTIONS OVER ERTS OR OPERS. STUDY DIRECTION IN WHICH OPERATIONAL LAND USE INVENTORY AND EARTH RESOURCE MGMT PROGRAMS SHOULD PROCEED.	PERFORM RESEARCH IN THE PRIORITY PROBLEMS OF OCEANOGRAPHY AND METEOROLOGY, ESPECIALLY THOSE ASSOCAITED WITH AN IMPROVED DATA BASE FOR LONG RANGE WEATHER FORECASTING AND FOR OCEAN RESOURCE MODELING.	DEVELOP ALL WEATHER CAPABILITY FOR BOTH ATMOSPHERIC STRUCTURE DETERMINATION AND SURFACE OBSERVATION.	VERIFY EOS COMPATIBILITY WITH THE SHUTTLE CAPABILITY FOR LAUNCH RESUPPLY AND RETRIEVAL. FINAL "SHAKE-DOWN" OF COMBINED SHUTTLE, THE RESUPPLIABLE OBSERVATORY AND THE FLIGHT SUPPORT SYSTEM.	DEVELOP REMOTE SENSING TECHNOLOGY FOR MEASUREMENT OF EARTH'S TRANSIENT ENVIRONMENT FROM SYNCHRONOUS ALTITUDE	INVESTIGATE FLARES AND RELATED PHENOMENA AND THEIR EFFECTS ON THE SOLAR-TERRESTRIAL SYSTEM THROUGH A WELL COORDINATED SET OF UNIQUE INSTRUMENTATION FOR OBSERVING TRANSIENT ULTRAVIOLET, HIGH-ENERGY AND VISIBLE RADIATION.	GLOBAL SCALE MONITORING OF WIDE RANGE OF PHYSICAL OCEAN PHENOMENA; SEA STATE CURRENTS, CIRCULATION, TIDES, WIND STRESS AND GEOID UNDULATIONS. DEMONSTRATE KEY FEATURES OF OPERATIONAL SYSTEM.	GLOBAL SCALE MONITORING OF WIDE RANGE OF PHYSICAL OCEAN PHENOMENA; SEA STATE CURRENTS, CIRCULATION, TIDES, WIND STRESS AND GEOID UNDULATIONS.	GLOBAL SCALE MONITORING OF THE EARTH'S SURFACE ON A PROTOTYPE OPERATIONAL BASIS. INVESTIGATE UTILITY OF THERMAL IR DATA FOR THE MONITORING OF EARTH'S RESOURCES. INVESTIGATE USE OF OFFSET POINTING SENSORS.
PAYLOAD (WIDTH x HEIGHT x LENGTH; WEIGHT) [FT x FT x FT; LBS]		THEMATIC MAPPER (3 x 3 x 7; ≤ 600) HIGH RESOLUTION POINTABLE IMAGER (3 x 3 x 7; ≤ 600) DATA COLLECTION SYSTEM (1 x 1 x 2; ≤ 77)	PASSIVE MULTICHANNEL MICROWAVE RADIOMETER (*79 FT^3; 200) ADVANCED ATMOSPHERIC SOUNDER (1.5 x 2.3; 100) RADIOMETRIC SCATTEROMETER (ELECTRONICS: 1 x 1 x 1 ANTENNA: 3.28 dia x 1.64; Total wts = 115) LIMB ATMOSPHERIC COMPOSITION PROFILIER (1.33 dia x 5 1/4; 168) ADVANCED MAPS (1.2 x 0.75 x 0.67; 80) OCEAN SCANNING SPECTROPHOTOMETER (2.16 x 1.46 x 0.75; 60)	SYNTHETIC APERTURE RADAR (ANTENNA: 27 x 2.5 x 1; ELECTRONICS: 5.1 FT^3; TOTAL WEIGHT = 387 LBS) THEMATIC MAPPER (3 x 3 x 7, ≤ 600)	* ENGINEERING MODEL HARDWARE OR BACKUP PAYLOADS FOR EOS-A.	4.9 FT DIAMETER (CASSAGRAIN) TELESCOPE ASSY. WITH VISIBLE, NEAR IR & THERMAL IR DETECTORS. RESOLUTION BETTER THAN 100 m VISIBLE AND NEAR IR, APPROX. 1000 m IN THERMAL IR. (TELESCOPE: 6.55 dia. x 13.1; 1144) (SENSOR ASSY: 6.73 dia x 3.3; 320) DATA COLLECTION SYSTEM (ANTENNA: 1.24 x 1.24 x 1; 31) (ELECTRONICS VOL: 1.32 FT^3; 44)	MINIMUM PAYLOAD: UV MAGNETOGRAPH (0.58x0.84x6; 100) EUV SPECTROMETER (0.84x0.84x6; 100) HIGH RESOLUTION X-RAY SPECTROMETER (0.54 x 0.84 x 6.5; 100) HARD X-RAY IMAGING (0.5x0.42x6.5,100) LOW/MEDIUM X-RAY POLARIMETER (0.67 x 0.67 x 3; 16) GAMMA RAY DETECTOR (1.5x1.5x3; 200) H-ALPHA PHOTOMETER (0.33x0.33x3; 20) FLARE FINDER (0.33 x 0.33 x 6; 30) ADDITIONAL SENSORS: IN ORDER OF IMPORTANCE ARE: HARD X-RAY SPECTRO. (1 x 1 x 3; 70) SOLID STATE X-RAY DETEC. (1x1x1; 20) CORONOGRAPH (0.42 x 1 x 6; 100) UV SPECTROMETER (0.67 x 1 x 6; 110) NEUTRON DETECTOR (0.83x1.67x3; 205)	RADAR ALTIMETER (___ x ___ x ___; 97) 5 CHANNEL MICROWAVE SCANNING RADIOMET. (___ x ___ x ___; 82) DUAL FREQUENCY SCATTEROMETER (___ x ___ x ___, 125) VISIBLE & SCANNING RADIOMETER (___ x ___ x ___, 20) SAR	ALTIMETER – K-BAND (0.66 x 0.66 x 1.64; 100) SCATTEROMETER – K-BAND (3.6 x 4.92 x 3.28; 200) IR SCANNER (3.28 x 3.28 x 3.96; 95) SATELLITE TO GROUND TRANSPONDER (0.82 x 0.66 x 0.66; 17.6) SATEL.-TO-SATEL. TRANSPONDER (1 x 2 x 2; 88) RETRO REFLECTORS (2 x 0.1 x 1.5; 44) COHERENT RADAR ALTIMETER (1 x 3.28 x 3.28; 161) *SAR	*5-BAND MSS (SCANNER: 1.77 x 1.95 x 4.2; 127) (MULTIPLEXER: 0.33x0.5x0.54; 7.5) HIGH RESOLUTION POINTABLE IMAGER (LIMITED CAPABILITY – SIZE, WEIGHT ARE UNKNOWN) DATA COLLECTION SYSTEM (ELECTRONICS: 0.5 x 0.17 x 0.54; 2.9) (ANTENNA: 1.1 x 1.5 x 0.85; 4.9)
ORBIT	ALTITUDE	418 nm	450 nm	418 nm	300 nm	19,323 nm	285 nm	430 nm	324 nm	500 nm
ORBIT	INCLINATION	98.5° SUN SYNCHRONOUS	98.76° SUN SYNCHRONOUS	98.5° SUN SYNCHRONOUS	28.5°	2° GEOSYNCHRONOUS	30°	108 DEG	90°	99° SUN SYNCHRONOUS
ORBIT	ASC. NODE TM	2330	1200	2330	NOT CRITICAL	N/A – POSITIONED AT 96° W. LONGITUDE	N/A	N/A	N/A	2330
POWER	TYPE	ONE SINGLE AXIS ORIENTED SOLAR ARRAY	ONE SINGLE AXIS ORIENTED SOLAR ARRAY	ONE SINGLE AXIS ORIENTED SOLAR ARRAY	ONE SINGLE AXIS ORIENTED SOLAR ARRAY OR BATTERY POWER	DUAL SINGLE AXIS ORIENTED SOLAR ARRAY	DUAL FIXED ARRAY	ORIENTED SOLAR ARRAY (TWO AXIS)	FIXED BODY MOUNTED SOLAR CELLS, OR SINGLE AXIS ARRAY + FIXED ARRAY, OR 2 AXIS ORIENTED ARRAY	ONE SINGLE AXIS ORIENTED SOLAR ARRAY
POWER	POWER LEVEL	500 w AVERAGE	550 w AVERAGE	** 450 w AVERAGE	500 w AVERAGE	400 w AVERAGE	235 w AVERAGE	465 w	375 w AVERAGE	270 w AVERAGE
ACS	REFERENCE	STELLAR	STELLAR	STELLAR	STELLAR	STELLAR	SOLAR/STELLAR	EARTH	EARTH	EARTH OR STELLAR
ACS	TYPE	3 AXIS, ZERO MOMENTUM	3 AXIS, ZERO MOMENTUM	3 AXIS, ZERO MOMENTUM	3 AXIS, ZERO MOMENTUM	3 AXIS, ZERO MOMENTUM	* 3 AXIS, ZERO MOMENTUM	3 AXIS, ZERO MOMENTUM	3 AXIS GRA GRAD. WITH MOMENTUM WHEEL SUN & HORIZON SENSORS	3 AXIS, ZERO MOMENTUM
ACS	ACCUR. POINT	0.007 DEG	0.05 DEG	0.05 DEG	0.007 DEG	0.017 DEG	1.2 SEC WITH FLARE FINDER; 1 MIN W/OUT	0.5 DEG	2 DEG	0.05 DEG
ACS	ACCUR. RATE	5×10^{-5} DEG/SEC	6.7×10^{-4} DEG/SEC	3.6×10^{-4} DEG/SEC	5×10^{-5} DEG/SEC	* 10^{-5} DEG/SEC	1 SEC OVER 5 MIN WITH FLARE FINDER	—	0.002 DEG/SEC	2×10^{-2} DEG/SEC
ACS	ACCUR. KNOW.	0.007 DEG	0.05 DEG	0.05 DEG	0.007 DEG	0.0017 DEG	< 5 SEC FROM FLARE FINDER	0.2 DEG	0.1 DEG	0.05 DEG
WIDE-BAND DATA	RATE	120 Mbps TWO 240 MBPS CHANNELS	2.5 MBPS	TWO 120 MBPS CHANNELS	NO	10 MBPS	128 KBPS	**1.6 MBPS	90 MBPS	TWO 15 MBPS CHANNELS
WIDE-BAND DATA	ON-BOARD STORAGE	YES (OR TDRS) YES (OR TDRS)	YES (OR TDRS)	YES (OR TDRS)	TELEMETRY ONLY	NO	YES (OR TDRS)	YES	YES	YES
LAUNCH VEHICLE		*DELTA TITAN	** DELTA TITAN	* DELTA TITAN	SHUTTLE-DIRECT	SHUTTLE	DELTA	DELTA	SHUTTLE-DIRECT	DELTA
TUG	TYPE	NOT REQUIRED INTEGRAL	NOT REQUIRED INTEGRAL	NOT REQUIRED INTEGRAL	NOT REQUIRED	NON INTEGRAL SPACE TUG	NOT REQUIRED	NOT REQUIRED	NO REQUIRED	NO REQUIRED
TUG	PROP TYPE	— HYDRAZ/SOLIDS	— HYDRAZ/SOLIDS	— HYDRAZ/SOLIDS	—	HYDRAZINE	—	—	—	—
TUG	NEED FOR ORBIT ADJ.	YES YES	NO NO	YES YES	NO	YES	NO	NO	NO	YES
SPACECRAFT CHARACTERISTICS	*WEIGHT – ON ORBIT	2500 LBS 4000 LBS	2400 LBS 3500 LBS	2500 LBS 4000 LBS	6500 LBS	2716 LBS	*** 2534 LBS	2050	2230 LBS	2000–2500 LBS.
SPACECRAFT CHARACTERISTICS	LENGTH	16 FT 27 FT	18 FT 27 FT	18 FT 21 FT	27 FT	22.7 FT	** 14 FT		15 FT	16 FT
SPACECRAFT CHARACTERISTICS	DIAMETER	7 FT 9 FT	7 FT 9 FT	7 FT 9 FT	9 FT	10.7 FT	** 7 FT		13 FT	7 FT
LAUNCH DATES		LATE 1978 / EARLY 1979	1980	1981	1980	1981	1978	1977	1982	1978
LIFETIME	LIFETIME	2 YEARS	2 YEARS	2 YEARS	7 DAYS	2 YEARS	1 YEAR	1 YEAR	5 YEARS	1 YEAR
LIFETIME	RETRIEVE	NO RETRIEVE RETRIEVE/REFURBISH 1983	NO RETRIEVE RETRIEVE/REFURBISH 1983	NO RETRIEVE RETRIEVE/REFURB. 1983	RETRIEVE IS PART OF MISSION	REFURBISH EVERY TWO YEARS	RETRIEVE/REFURBISH EVERY 2 YEARS	NO RETRIEVE	NO RETRIEVE	NO RETRIEVE
NOTES	*NOT INCLUDING PROPULSION SYSTEM WEIGHTS	*DELTA CONFIGURATION WILL CARRY ONLY THE THEMATIC MAPPER & DCS	*INCLUDES TWO 3.6 FT DIAMETER SCANNING ANTENNAS. **DELTA CONFIGURATION WILL CARRY THE PMMR, AS MANY OTHER INSTRUMENTS AS POSSIBLE BUT NOT THE RADIOMETER SCATTEROMETER.	*DELTA CONFIGURATION WILL CARRY ONLY THE SAR **1250 WATTS PEAK POWER FOR 10 MIN REQUIRED FOR THE SAR.	*ASSUME PAYLOADS TO BE TURNED ON AND IMAGERY ACQUIRED TO SUPPORT FULL UP SHUTTLE AND SPACECRAFT COMPATIBILITY TEST MISSION. MORE LIKELY MISSION WILL HAVE MINIMUM S/C CAPABILITY (NO OPERATING PAYLOADS, NO SOLAR ARRAY)	SYSTEM MUST BE CAPABLE OF PROVIDING TEN 70 MILIRADION (4°) SCANS PER HOUR TO SCAN THE EARTH AT CONSTANT SUN ELEVATION.	*10 MIN SLEW IN ONE AXIS FOLLOWED BY 10 MIN SLEW IN SECOND AXIS IN 30 SECONDS TOTAL IS REQUIRED. SLEW BASED ON ERROR SIGNALS FROM FLARE FINDER SENSOR. **END VIEWING CONFIGURATION. ***WEIGHT CORRESPONDS TO MINIMUM PAYLOAD.	*POTENTIAL ADDITION OF SAR COULD HAVE MAJOR IMPACT ON SENSOR COMPLIMENT AND SPACECRAFT CONFIGURATION. **DOES NOT INCLUDE PROVISIONS FOR SAR.	*POTENTIAL ADDITION OF SAR COULD HAVE MAJOR IMPACT ON SENSOR COMPLIMENT AND SPACECRAFT CONFIGURATION.	*SYSTEM MAKES EARLY USE OF EOS HARDWARE.
REFERENCES		MULTIPLE DOCUMENTS SUPPLIED WITH THE RFP	INTERNAL GE CONCEPTUAL EOS-B PAYLOAD	INTERNAL GE CONCEPTUAL EOS-C PAYLOAD	"EOS REQUIREMENTS FOR EARLY SHUTTLE FLIGHTS", GSFC MAY 1973	"SSPD DATA SHEETS, EO-09-A" 10/9/73	"SOLAR MAXIMUM MISSION CONCEPTUAL STUDY REPORT X-073-74-42" GSFC, JAN 1974	"SEASAT TASK TEAM REPORT" "SEASAT SOURCE VERIFICATION STUDY," GE FINAL REPORT, 4/8/74	"SSPD DATA SHEETS OF-07-A" REV. 10/15/73	INTERNAL GE CONCEPTUAL PAYLOAD

FOLDOUT FRAME 1

FOLDOUT FRAME 2

EOS-C. An all-weather observatory with a Synthetic Aperture Radar as the primary payload sensor. A Thematic Mapper was added to provide correlative visible and IR data to support R&D goals of full spectrum data utilization.

EOS-D & E. Currently undefined payloads; payload definition to depend on results from the first three EOS missions.

Shuttle Resupply Demonstration Test Flight. Will serve as the first Space Shuttle checkout mission to verify the resupply/retrieve concept. An EOS spacecraft built from back-up hardware is planned to be the sample automatic payload during this test flight.

Other Missions

SEOS. A geosynchronous Earth viewing satellite.

Solar Max. An Earth orbiting solar viewing satellite, initially launched with a Delta and later retrieved with a Shuttle for reuse.

Seasat-A/B. Global ocean monitoring satellites. Seasat-A is Delta launched in 1977; Seasat-B is shuttle launched in the early 1980's.

5-Band MSS. An early EOS supported Earth Observation mission which is not constrained by sensor development. The payload to consist of a 5-Band MSS as currently defined in the ERTS program plus a "scaled down" HRPI with a narrow field of view and nominally two spectral bands to be consistent with the 5-Band MSS data rates ($\sim$15 Mbps). An alternate payload considered is two 5-Band MSS' viewing adjacent 100 nm swaths.

The orbit tradeoff parameters that constrain the orbit selection for EOS-A are presented in Table 2-2 showing the orbit tradeoff parameter, the significance of the parameter for EOS-A and the nominal EOS-A requirements. The parameters that drive the orbit selection for EOS-A are the sidelap of 5 to 15%, repeat cycle of 15 to 18 days, access time of 2 to 4 days and altitude range of 300 to 500 nm.

2.3 LAUNCH VEHICLE CONSTRAINTS

The constraints placed on the orbit selection by the launch vehicle can be separated into two areas, constraints from expendable launch vehicles and constraints imposed by the ultimate use of space shuttle. The expendable launch vehicle and the Space Shuttle allowable delivery

Table 2-2. Orbit Tradeoff Parameters

Parameters	Significance For EOS-A	Nominal EOS Requirements
Coverage	Sun-sync orbits allow view to 81° lattitude; coverage determined by swath width & altitude (period).	Global
Swath Width	Key in sizing instrument optics & detectors.	185 Km (100 nm)
Sidelap	Tradeoff between ACS accuracy and orbit maintenance vs. excessive data.	5-15%
Sun Synchronous	Minimizes illumination changes on ground scene; simplifies power and thermal subsystems.	Sun Synchronous
Descending Node Time	Affects illumination of scene, thermal and power S/S; selected independent of other variables.	9:30 a.m. to 2:30 p.m.
Repeat Cycle	Provides periodic data gathering and simplifies flights ops and and data procurement.	15 to 18 Days
Access Time	Dictates HRPI offset pointing.	2 to 4 Days
Interlace Pattern	Determine time between imaging of adjacent swaths, best mosaicing potential with minimum time between adjacent swaths.	Equal to the access time
Altitude	Determine repeat cycle and interlace pattern. Instrument optics, L/V payload capability and integral propulsion system requirements.	300 to 500 nm (555 to 927 Km)

capabilities as a function of mission altitude are presented in Section 4.0 - Launch System Parametric Performance Analysis. The other constraints imposed by the launch vehicles are discussed below.

The alternate launch vehicles specified in the RFP were Delta 2910, Titan IIID NUS and Space Shuttle. The Titan IIIB NUS has been added by General Electric to include a relatively inexpensive launch vehicle with considerably improved performance over the other low cost expendable launch vehicle (Delta 2910). The Delta 3910 has also been added since its projected performance and cost appear attractive for early EOS flights.

Figure 2-1 graphically illustrates the relative shroud restrictions for a Delta or a Titan launch. The Delta shroud has an internal envelope 86 inches in diameter with a cylindrical length of approximately 177 inches while the Titan shroud for WTR launch allows an internal diameter varying from 111.4 inches to 107.7 inches with a cylindrical length exceeding 400 inches. The shroud restrictions on Delta are the most severe constraint imposed by a Delta launch.

The space Shuttle payload envelope is 180 inches in diameter and 720 inches long. It should be noted that OMS kits required for EOS orbits must be subtracted from this payload envelope along with auxiliary Shuttle equipment such as the flight support system and the module exchange mechanism (if used).

The launch vehicle injection errors for the alternate vehicles under consideration are presented in Table 2-3. The large deviation in altitude for Delta 2910 requires significant ΔV at orbit injection to establish the precise orbits required for EOS-A (HRPI and TM). The 0.2 degree inclination error for Titan IIID has two effects. First, it causes a change in the position of the descending node which can be compensated by a very small altitude correction (as was done with ERTS-1). Second, it changes the time of the decending node. The error in node time would accumulate to about half an hour after two years for a 0.2 degree (3σ) inclination error. If correction is desired it can be performed in conjunction with the orbit circularization required if a Titan IIID places the spacecraft into an elliptical orbit. The Delta launch vehicle uses an inertial guidance system while the Titan launch vehicles use radio guidance which is compatible with the EOS orbits. Titan can also be configured to obtain guidance from the spacecraft which is not recommended for EOS applications due to interfacing complexity.

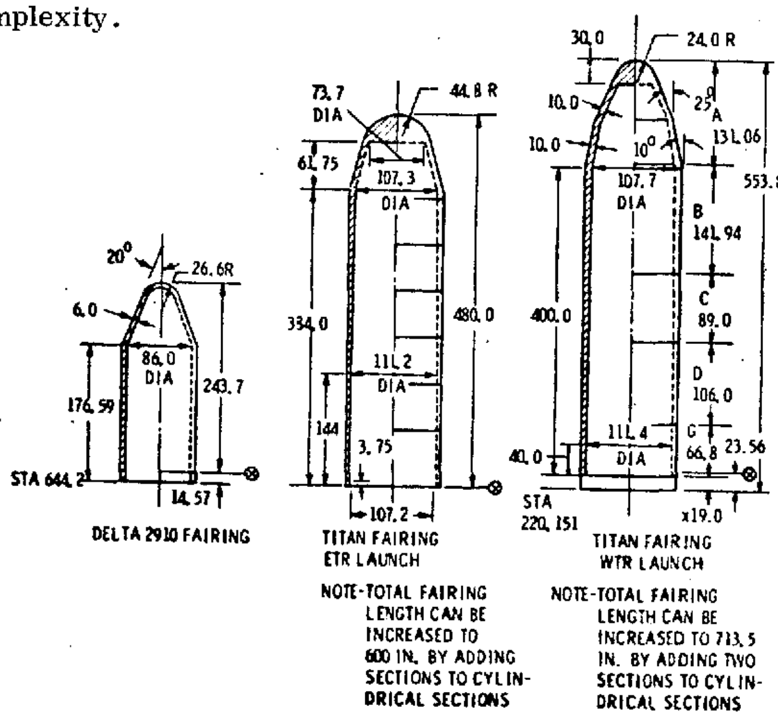


Figure 2-1. Launch Vehicle Shroud Envelopes

Qualification test acceleration levels for steady state, and sinusoidal and random vibrations and acoustic noise levels are given in Table 2-4 for the range of EOS launch vehicles. These qualification test levels are 150% of the expected flight levels.

The final and possibly the most important constraint/requirement from the launch vehicle is the launch vehicle cost. The cost figures used in the launch vehicle/orbit evaluation are presented in Table 2-5. The Titan costs have been increased to include a non-Air Force user charge.

Launch Vehicle	Injection Errors 3σ	
	Deviation In Altitude (NM)	Deviation In Inclination (Deg)
Delta 2910	± 14	± 0.04
Titan III B/NUS	± 0.6	± 0.10
Titan III D/NUS	± 1.0	± 0.20
Shuttle	± 3.0	± 0.05

Table 2-3. Launch Vehicle Injection Errors

The compatibility with Shuttle of the spacecraft and orbit selection for EOS is of prime importance. Orbits selected for EOS must be compatible with Shuttle retrieval (either directly or thru the use of an integral propulsion system on the spacecraft). A relative cost analysis of direct Shuttle access to the mission orbit or retrieval at a lower altitude using the integral propulsion system has been performed to establish the recommended approach. This is described in Section 5.

2.4 INSTRUMENT CONSTRAINTS

The Instrument Point Design Studies for the three candidate Thematic Mappers, the three candidate HRPI's and the one pushbroom array HRPI were all generated to slightly different baselines as shown in Table 2-6. The general guidelines to which the instrument contractors were working for the August 1973 parametric point studies were:

	TM	HRPI
Weight	600 lb	600 lb
Altitude	300-500 nm	300-500 nm
Size	84" x 36" x 36"	84" x 36" x 36"
Power	100 w	100 w

Table 2-4. EOS Spacecraft Qualification Test Levels

A. Sinusoidal Vibration

Launch Vehicle / Axis	Thor/Delta		Titan IIID		Space Shuttle	
	Frequency (Hz)	Acceleration (G, 0-pk)	Frequency (Hz)	Acceleration (G, 0-pk)	Frequency (Hz)	Acceleration (G, 0-pk)
Longitudinal Axis	5 - 15* 15 - 21 21 - 100	2.3 6.0 2.3	5 - 20* 20 - 50 50 - 200	9.0 in/sec 3.0 in/sec 2.3 in/sec		TBD
Lateral Axis	5 - 14* 14 - 100	2.0 1.5	5 - 22* 22 - 200	2.0 in/sec 1.5 in/sec		TBD

Sweep Rate: 2 octaves/minute

* Limited with the performance of the exciter. The amplitude in these frequency ranges shall not exceed 0.5 inches D.A.

B. Random Vibrations (Thrust and Lateral Axes)

	Frequency (Hz)	PSD (G^2/Hz)	GRMS	Time (sec/Axis)
Thor Delta	20 - 300 300 - 700 700 - 2000	+4 dB/Oct .16 -3 dB/Oct	14.1	20
	20 - 300 300 - 700 700 - 2000	+4 dB/Oct .07 -3 dB/Oct	9.5	70
Titan IIID	20 - 250 250 - 2000	6 dB/Oct .16	17.0	240
Shuttle	20 - 100 100 - 250 250 - 2000	+6 dB/Oct .65 -6 dB/Oct	24.3	90

Table 2-4. EOS Spacecraft Qualification Test Levels (Continued)

C. Quasi Steady Acceleration, G's

Launch Vehicle & Condition	Longitudinal (g)	Lateral (g)
Thor Delta		
Max. Lateral (Lift-off)	- 4.4	± 3.0
Max. Compression (MECO/POGO)	- 18.0	± 1.0
Max. Tension	1.5	± 3.0
Titan IIIB NUS		
Max. Lateral (Lift-off)	- 2.9	± 2.5
Max. Compression (Stage II shutdown)	- 13.5	± 1.3
Max. Tension (Stage I shutdown)	3.1	± 1.9
Titan IIID NUS		
Max. Lateral (Lift-off)	- 3.2	± 3.2
Max. Compression (Stage I shutdown)	- 9.5	± 1.9
Max. Tension (Stage I shutdown)	3.2	± 1.9
Shuttle		
Lift-off	- 3.5	± 1.3
Orbiter end burn	- 5.0	± .6
Entry	0.4	± 4.5
Landing & Braking	± 2.3	3.8
Crash (ultimate applied separately	9.0	4.5
	- 1.5	- 2.0

D. Acoustic Noise

Octave Band Center Frequency (Hz)	Sound Pressure Level: dB ref. .0002 dynes/cm^2			
	Thor/Delta		Titan IIID	Shuttle
31.5	129	124	124	131
63	130	125	130	137
125	134	129	138	141
250	139	134	143	143
500	147	142	142	143
1000	141	136	137	141
2000	138	133	133	138
4000	131	126	130	134
8000	128	123	128	130
Overall	149	144	147	149
Duration (sec)	20	70	120	120

2.4.1 SIZE, WEIGHT, POWER

The resulting design parameters are shown in Table 2-7 where the results are actually more representative of degree of design completeness than of basic differences between approaches. Thus, GE assumed reasonable latitude in these instrument parameters considering the following:

Table 2-5. Launch Vehicle Costs (1976 Dollars)

Launch Vehicle	Recurring Cost
Delta 2910	6M
Delta 39 10	8M
Titan IIIB NUS	12.2M
Titan IIID NUS	25 to 44M
Shuttle	9.8M (max. round trip)

- o The object plane scanner (Hughes) should theorically be slightly smaller and lighter than image plane scanners but not by the differences shown in Table 2-7. The Hughes update designs were responsive to the need to reduce weight, envelope and power for Delta launch compatibility. The design is based on both the MSS and VISSR instruments and therefore the weight and power parameters are based on fairly detailed designs (but probably 10 to 20% optimistic).
- o The length of the Te design cannot be reduced below six feet due to the physical size of the roof wheel and optical path length requirements for the EOS-A baseline. The weight and power can both be reduced with sufficient emphasis.
- o Honeywell's point study was not in sufficient detail to provide realistic scrubbed estimates of size, weight or power. Their solution to the thermal stability problem is to use full power throughout the orbit. Significant improvements in weight and power are realistic to assume.

2.4.2 ORIENTATION

Table 2-7 also indicates the orientation of the candidate instruments to the spacecraft velocity vector. The Honeywell design can be reconfigured fairly easily for either orientation. In the Te TM and ScHRPI designs, the optical axis should be perpendicular to the spacecraft velocity vector. This allows the roof wheel to be mounted vertically within the

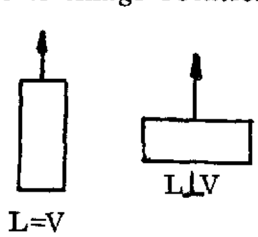
Table 2-6. Summary of Instrument Contractors Point Design Parameters

	TM		HRPI	
	Original	Update		
Altitude, Km				
Honeywell	914	900	716	
Hughes	914	717	717	
Te-Gulton	914	715	715	
Westinghouse	N/A	N/A	914	
Angular IFOV, μ rad				
Honeywell	30	33	14	
Hughes	30	30	14	
Te-Gulton	33	35	14	
Westinghouse	N/A	N/A	10.9	
Descending Node Time				
Honeywell	9:30	9:30	9:30	
Hughes	9:30	9:30	TBD	
Te-Gulton	9:30	11:30	11:30	
Westinghouse	N/A	N/A	9:30	
(Sc) HRPI Offset Pointing Angle			Original	Update
Honeywell			±40	±30 preferred
Hughes			±40	
Te-Gulton			±45	
Westinghouse			±10	± ?

Table 2-7. Instrument Design Physical Parameters

	TM Original	TM Update	HRPI
Size			
Honeywell	83L x 36D in.	72L x 36D in.	72L x 36D in.
Hughes	83 x 44 x 25 in.	a) 67 x 36 x 20 in. b) 42 x 36 x 36 in.	TBD
Te-Gulton	84 x 36 x 38 in.	84 x 36 x 38 in.	84 x 36 x 38 in.
Westinghouse	N/A	N/A	72-1/2 x 27D in.
Weight, lbs.			
Honeywell	450	600	600
Hughes	401	a & b) 320	318
Te-Gulton	598	598	598
Westinghouse	N/A	N/A	553
Power, watts			
Honeywell	180 + 50 Heat	180 + 50 Heat	180 +80 point +50 heat
Hughes	55	45	81 + 10 Cmd Func.
Te-Gulton	110 + 10 Heat	110 + 10 Heat	100 + 50 heat
Westinghouse	N/A	N/A	100 +23 point +21 heat
Orientation			
Honeywell	L=V	Either	L=V
Hughes	L⊥V	a) L⊥V b) L=V	L=V
Te-Gulton	L⊥V	L⊥V	L⊥V
Westinghouse	N/A	N/A	L=V*

* Reconsidered due to image rotation; now L must be ⊥V



instrument (aids ground testing) and makes offset pointing less of a problem. Hughes has distinct designs for the TM and ScHRPI. The TM could be redesigned to accommodate either orientation, however, the orientation shown would be more cost effective due to the commonality with the MSS design. Originally, the HRPI was oriented with the optical axis parallel to the velocity vector, but this would cause an image rotation with respect to the detector array as a function of offset pointing angle. Therefore, the present design requires that the optical axis be mounted perpendicular to the velocity vector.

2.4.3 MOUNTING

The HAC TM, HRC TM, Te TM and ScHRPI, and the Westinghouse HRPI all are designed to use a three point, thermally isolated pickup through an as-yet undefined CG. The HAC and HRC ScHRPI's both have large rotating sections mounted off smaller stationary pieces and the mounting techniques have not been suitably documented by the contractors.

2.4.4 FIELDS OF VIEW - OPTICAL

The TM's require approximately $\pm 8^{\circ}$ field of view from spacecraft Nadir along the instrument's optical axis. The HRPI and ScHRPI's require up to a $\pm 48^{\circ}$ clear field of view. These can be accommodated by mounting the ScHRPI more earthward within its instrument module than the TM.

2.4.5 FIELD OF VIEW - COOLERS

All candidate TM's require a radiant cooler for the thermal band detectors. In the point study reports, the contractors sized the coolers and oriented the fields of view for a 9:30 orbit. These designs will have to be modified by the contractors for an 11:30 orbit.

2.4.6 CONTAMINATION PROTECTION

Requirements for optical and cooler protection covers have not been worked yet.

2.4.7 LAUNCH MODES

Honeywell requires the scanner ON during launch due to bearing preload considerations. Neither Te nor Hughes has such a requirement.

2.5 SPACECRAFT CONSTRAINTS

2.5.1 INTRODUCTION

The spacecraft constrains the orbit/launch vehicle selection in a number of ways including spacecraft weight margin for alternate launch vehicles as a function of altitude, spacecraft packaging restrictions as a function of alternate shroud envelope and orbit and launch vehicle selection flexibility as a function of on-board propulsion system impacts. Preliminary spacecraft evaluations have been performed to establish the constraints which are presented in this section.

2.5.2 SPACECRAFT WEIGHT CONSTRAINTS

Five basic spacecraft arrangements have been synthesized to establish a range of weights for EOS-A. These spacecraft include three Delta and two Titan arrangements. The three Delta arrangements include maximum capability, nominal capability and light weight spacecraft with their weights shown in Table 2-8. The propulsion system (RCS, OA & OT) weights have not been included in the table since their weights are already factored into the launch system capabilities presented in Section 5.0.

The maximum capability Delta spacecraft includes added redundancy in C&DH, added redundancy and two 200 Mbps tape recorders in the wideband area and the maximum weight payload instruments giving a basic spacecraft weight of 1599 lbs, wideband weight of 576 lbs and a payload instrument weight of 1205 lbs. Total spacecraft weight is 3380 lbs. Launch of the maximum capability Delta spacecraft requires a Delta 3910 with the weight marginal at a mission altitude of 420 nm depending on retrieval altitude.

The nominal capability Delta spacecraft includes a non-redundant basic spacecraft weighing 1472 lbs and a wideband system with one 200 Mbps tape recorder weighing 343 lbs. The payload instruments include the Westinghouse HRPI, the Hughes TM and a DCS weighing 54 lbs. The nominal capability spacecraft weighs 2752 lbs which is well within the capability of Delta 3910 while exceeding the Delta 2910 capability to 420 nm by approximately 350 lbs.

Table 2-8. EOS-A Delta Spacecraft Weights (lbs)
(Minus Propulsion System Wts.)

	Max. Capability	Nominal Capability	Light Weight
Structure & Modules	715	605	555
ACS	100	100	100
Power	247	247	237
Solar Array & Drive	125	125	108
C&DH	102	85	85
Harness & Sig. Cond.	132	132	132
Thermal	95	95	95
Adapter	83	83	71
Total Basic S/C (Minus Pneumatics)	(1599)	(1472)	(1383)
Wideband Communication	576	343	160
Total Mission Pec. (Minus O.A. & O.T.)	(576)	(343)	(160)
HRPI	553	553	330
Thematic Mapper	598	330	330
DCS	54	54	12
Total Payload Instr.	(1205)	(937)	(672)
Total Spacecraft (Minus Propulsion)	(3380)	(2752)	(2215)

The light weight spacecraft is compatible with a Delta 2910 launch to 420 nm, and a Shuttle retrieval at 330 nm.

The basic spacecraft changes between the nominal capability spacecraft and the light-weight spacecraft include a 50 lb reduction in structural weight due to the significant weight reduction in the wideband system and the payload instruments plus weight reduction in the power system and solar array due to the deletion of the wideband tape recorder. The light-weight basic spacecraft also uses a lighter weight solar array design and a shorter adapter. Additional weight reductions in the order of 100 lbs can be made in the basic spacecraft weights, however, these reductions would come at the expense of increased cost. The significant weight reduction shown in the mission peculiar components is achieved by eliminating the wideband tape recorder. Real time coverage of the USA is maintained along with the low cost user link. A weight reduction of 265 lbs. is achieved in the payload instruments by substituting the Hughes HRPI for the Westinghouse HRPI and using the ERTS DCS system. It should be noted that preferenced is not being made to one instrument over the other but that the lower instrument weight is required for a light-weight spacecraft compatible with the Delta 2910. Additional weight savings are retrieved at the mission altitude (120 lbs) or no retrieve capability is provided (220 lbs). The light-weight spacecraft weighs 2215 lbs, giving a weight margin of 145 lbs.

The two Titan spacecraft configurations also shown in Table 2-9 include a nominal capability spacecraft (similar to the nominal Delta spacecraft) and a maximum capability spacecraft that includes resupply, added redundancy, a second wideband tape recorder and increased instrument weights. The nominal capability Titan spacecraft weighs 2971 lbs having a weight margin of 1204 lbs for a Titan IIIB integral tug launch to 420 nm orbit (assuming Shuttle retrieval at 250 nm).

Table 2-9. EOS-A Titan Spacecraft Weights (lbs)
(Minus Propulsion System Wts)

	Max. Capability	Nominal Capability
Structures & Modules	1115	715
ACS	100	100
Power	247	247
Solar Array & Drive	125	125
C&DH	102	85
Harness & Sig. Cond.	132	132
Thermal	95	95
Adapter	140	125
Total Basic Spacecraft (Minus Pneumatics)	(2123)	(1691)
Wideband Communication	576	343
Total Mission Peculiar (Minus D.A. & O.T.)	(576)	(343)
HRPI	553	553
Thematic Mapper	598	330
DCS	54	54
Total Payload Instr.	(1205)	(937)
Total Spacecraft (Minus Propulsion)	(3904)	(2971)

The maximum capability Titan spacecraft weighs 3904 lbs with the weight margin reduced to 271 lbs. This margin can be increased to 446 lbs if the Shuttle retrieval altitude is raised to 330 nm. These spacecraft weight constraints can be used in evaluating alternate launch vehicles and orbits for EOS-A by comparing the weights to the parametric performance data for the alternate launch systems. This data is presented in Section 5.0 as a function of mission altitude and Shuttle retrieval altitude.

2.5.3 INTEGRAL PROPULSION SYSTEM CONSTRAINTS

Propulsion system constraints which are discussed within this section assume use of a subsystem design utilizing a single propellant (hydrazine) for performing the functions of reaction control, orbit adjust and orbit transfer. This type of propulsion system design was selected based upon the trade results presented in the "Design Cost Tradeoff Studies;" Report #3.

The types of propulsion subsystem designs which are considered candidates for performing the EOS spacecraft functions are typically evaluated as to their constraints on the following areas of interest:

1. Selection of the launch vehicle,
2. Limitation upon the range of mission orbits, and
3. Compatibility with a Shuttle launch and retrieval.

The selected propulsion subsystem type offers minimal constraints in these areas because of its "building block flexibility" in the areas of propellant tankage size and the available range of hydrazine rocket engine thrust levels.

The selected propulsion system is able to accommodate the entire stable of available launch vehicles applicable to the EOS spacecraft mission. Payloads launched from vehicles having capability limitations to either low altitude circular orbits (below mission altitude) or to elliptical orbits can be transferred to the desired mission orbit with the integral propulsion system by using tankage with increased propellant capability and by the addition of orbit transfer rocket engines. These same additions accommodate the range of mission orbit altitudes (300 to 500 nm) and the range of Shuttle retrieval altitudes (200 to 350 nm). This component flexibility results in a propulsion system design that offers negligible constraint to the EOS missions within the areas previously defined.

2.5.4 SPACECRAFT CONFIGURATION CONSTRAINTS

Two basic Delta spacecraft configurations and two basic Titan spacecraft configurations have been established in order to develop configuration dependent constraints on the launch vehicle and orbit selection. The two basic spacecraft configurations for both the Delta and Titan include the GSFC baseline and an alternate configuration which is a variation of the GSFC baseline. Each of these configurations present varying constraints on the payload envelope, subsystem module envelope and orientation and propulsion module space allocation.

The GSFC Delta configuration is shown in Figure 2-2. The impact of the small Delta shroud on the spacecraft configuration can be seen from the figure. The subsystem modules must be packaged in a triangular arrangement (as shown in Section A-A) which necessitates alternate subsystem module orientations for the Delta and Titan spacecraft configurations. These alternate orientations impact the thermal design of the subsystem modules which are designed to fly on both Delta and Titan. The payload module is also constrained by the shroud to a length less than 100 inches unless the Delta shroud is elongated. This limitation impacts the variety of instruments that can be mounted in the instrument section. The interstage adapter transition ring shown causes a weight penalty of approximately 100 lbs over a conventional adapter which limits the instrument payload weight to less than 600 lbs for a Delta 2910 launch. This payload weight is insufficient for a TM and HRPI payload. The transition ring/interstage adapter trades are discussed in greater detail in Report #3.

The alternate Delta spacecraft configuration is shown in Figure 2-3. The most significant differences between the GSFC and alternate Delta configurations are:

- o Reduced subsystem module size from 48 x 48 x 20 inches to 40 x 48 x 16 inches which allows the same subsystem module arrangement for Delta and Titan configurations.
- o Use of a conventional aft adapter to replace the interstage adapter allowing an additional 100 lbs for payload instruments.

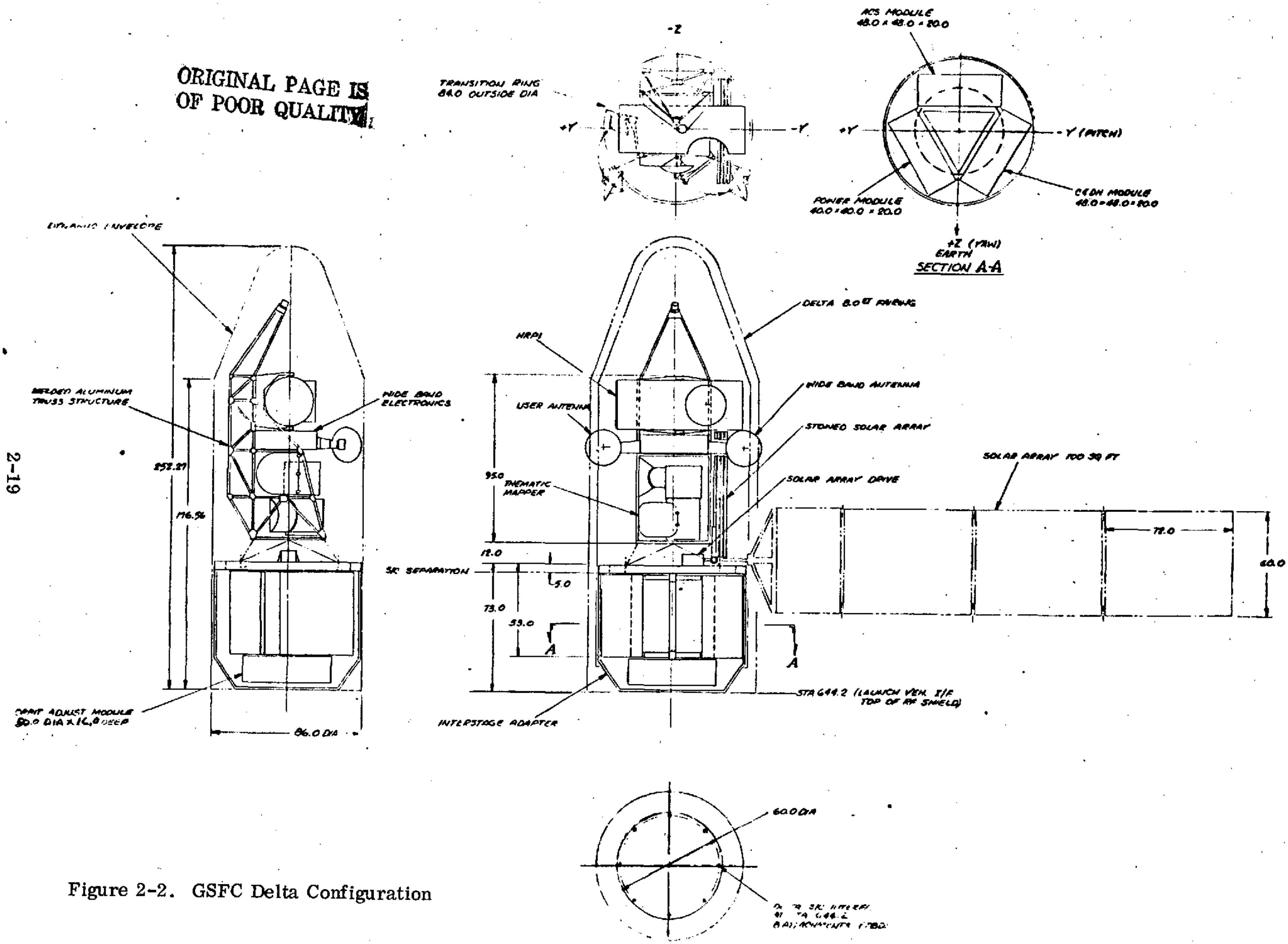


Figure 2-2. GSFC Delta Configuration

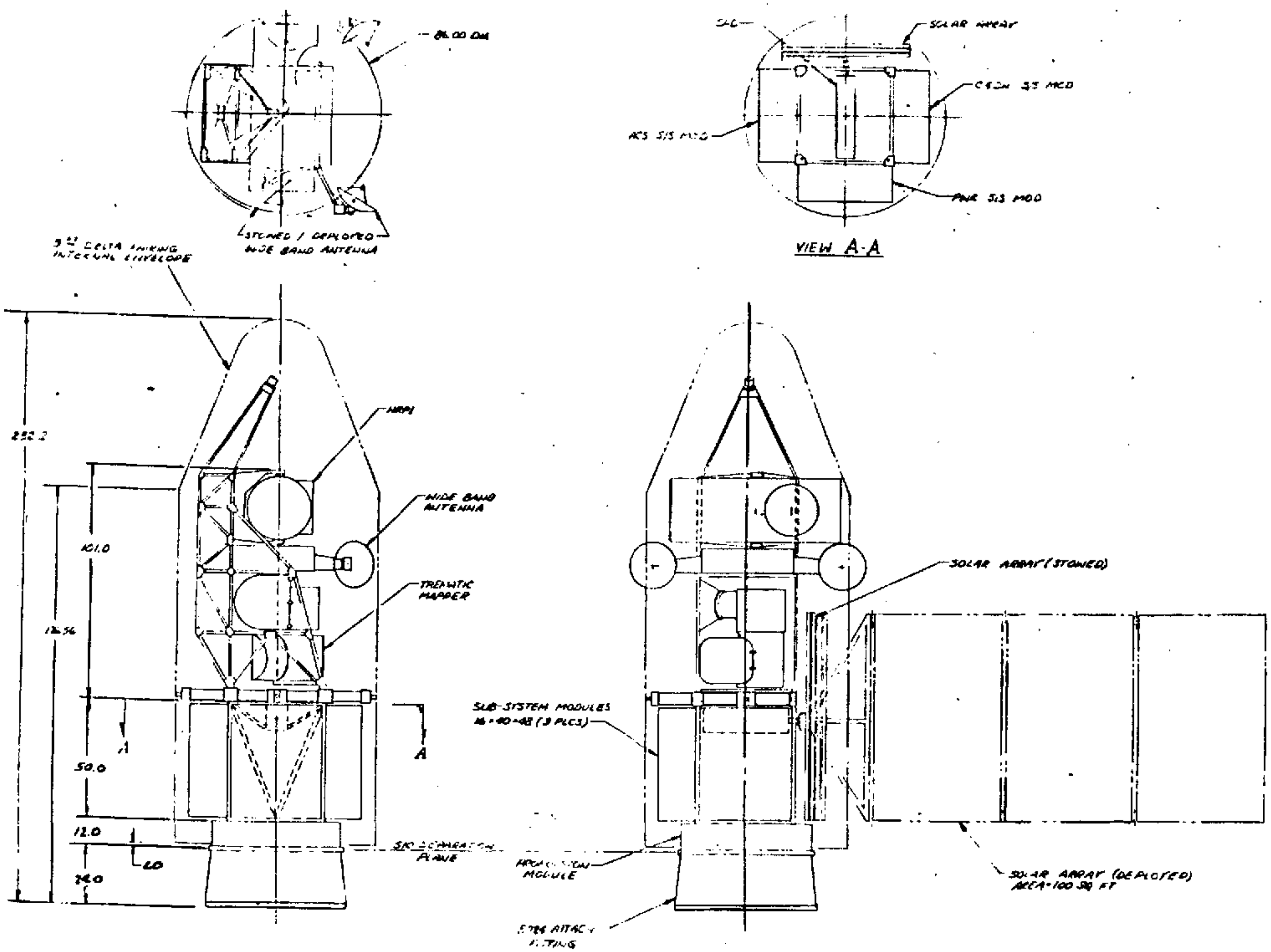


Figure 2-3. Alternate Delta Configuration

The Delta shroud length still restricts the allowable length and diameter of the payload instruments but the additional weight available due to the elimination of the interstage adapter and transition ring makes a Delta launch at 418 nm feasible if the instrument payload can be limited to less than 700 lbs. If the mission altitude is reduced below 418 nm the allowable instrument weight can be increased.

The GSFC Titan/Shuttle configuration which illustrates the advantages of the enlarged Titan shroud is shown in Figure 2-4. Ample volume exists within the instrument section to mount the entire range of HRPI and TM designs investigated in the point design studies by the instrument contractors. The Titan shroud volume and weight capability also makes the transition ring and interstage adapter concept more feasible although it is estimated that a weight penalty of approximately 350 lbs. is associated with the interstage adapter for Titan (see Report #3).

The alternate Titan/Shuttle configuration shown in Figure 2-5 provides similar benefits to the GSFC configuration allowing ample volume for packaging payload instruments which can be oriented as shown in the alternate design (parallel to the velocity vector), the GSFC design (perpendicular to the velocity vector) or one parallel and one perpendicular to the direction of flight. The alternate Titan configuration can be made compatible with either the interstage or conventional aft adapter.

One significant difference between the four spacecraft configurations is the allowable volume for mounting the propulsion system. The alternate space allocations are shown in Figure 2-6. The Titan baseline space allocation can be increased to be similar to the alternate Titan system if the instrument mounted in the subsystem section is deleted or moved forward of the transition ring. Either Delta system appears to have ample volume for packaging a Delta propulsion system.

2.6 EVALUATION CRITERIA

The evaluation criteria for the selection of the EOS-A launch system and orbit involves the following key elements that are summarized in Table 2-10.

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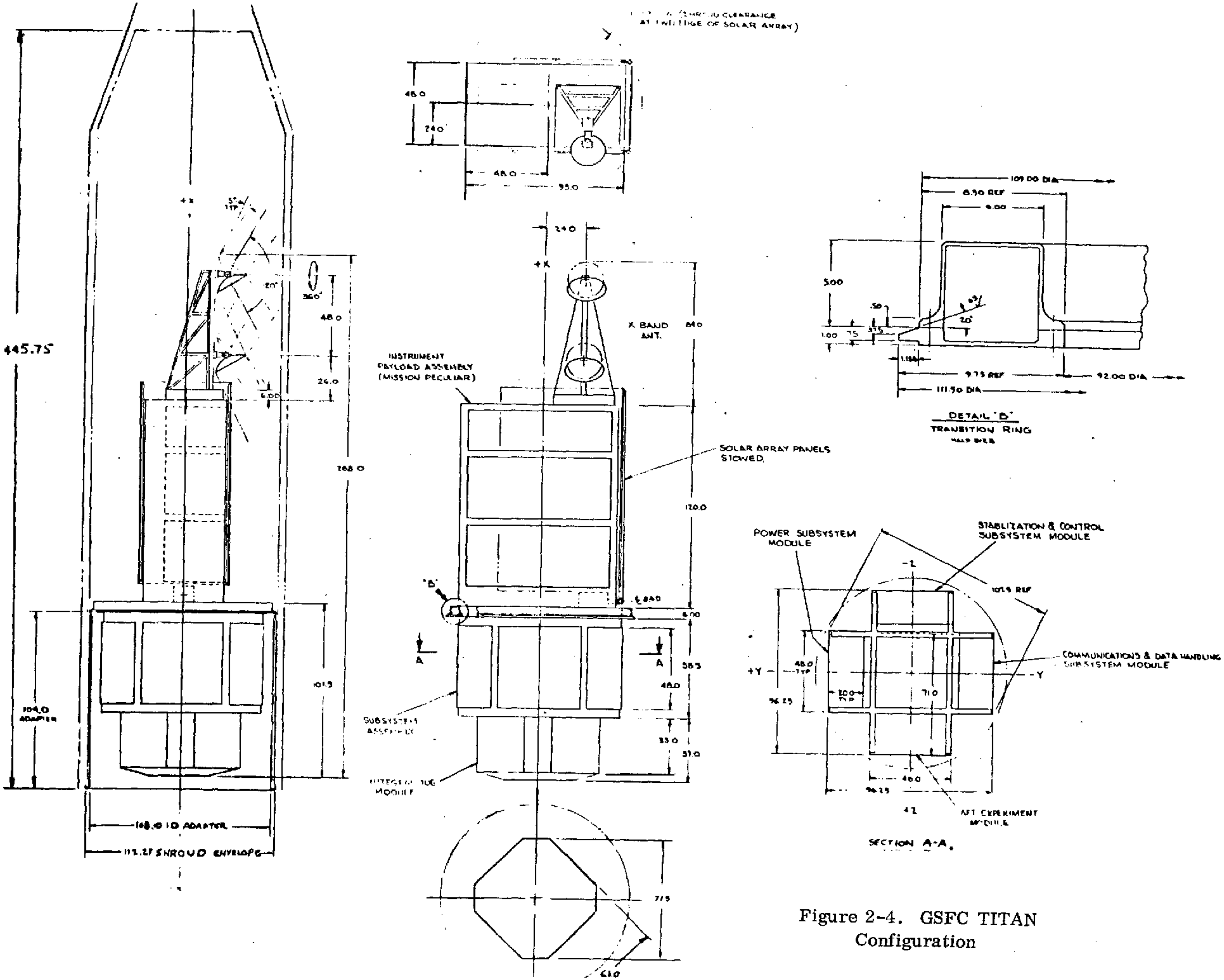

Figure 2-4. GSFC TITAN Configuration

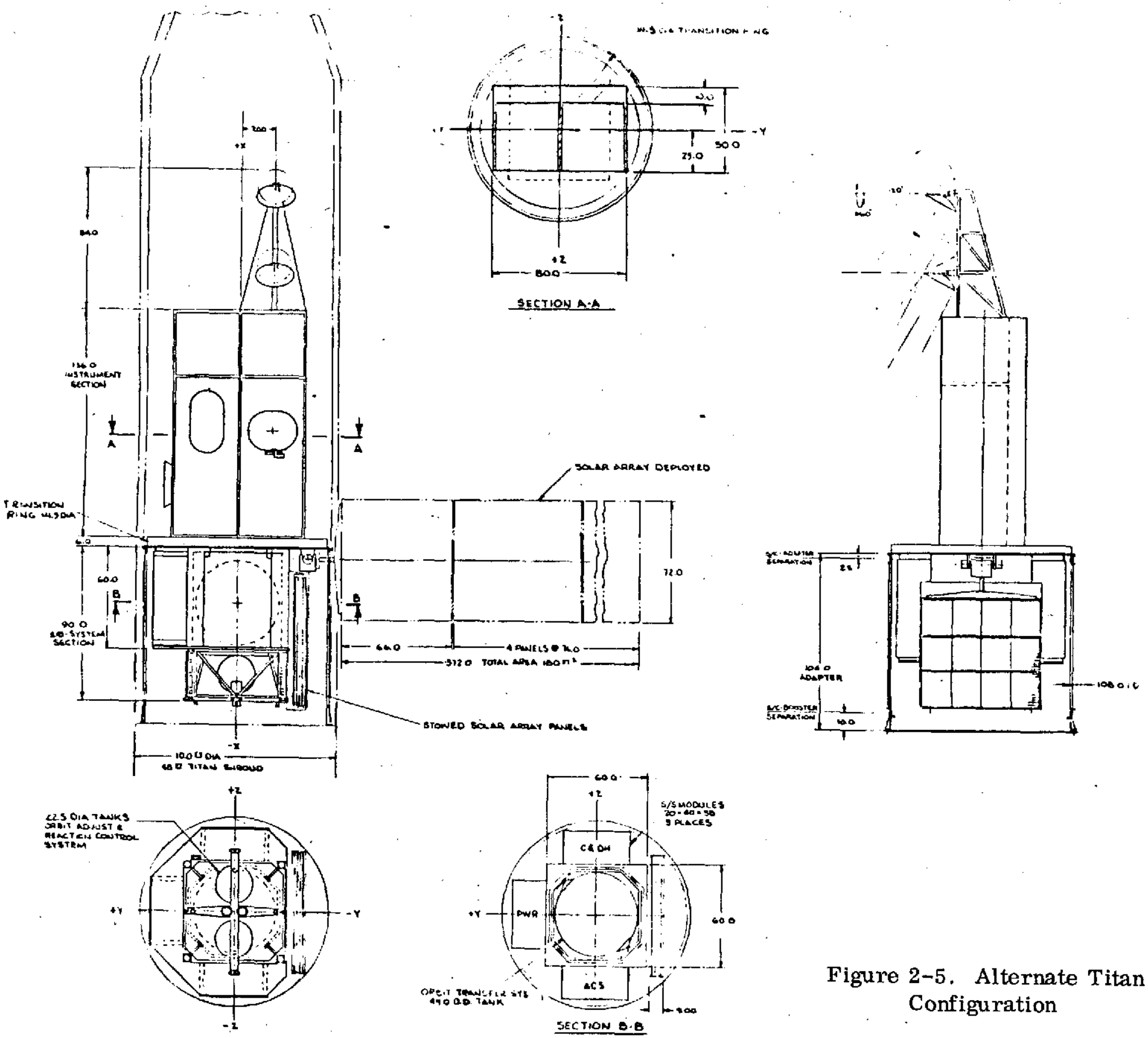

Figure 2-5. Alternate Titan Configuration

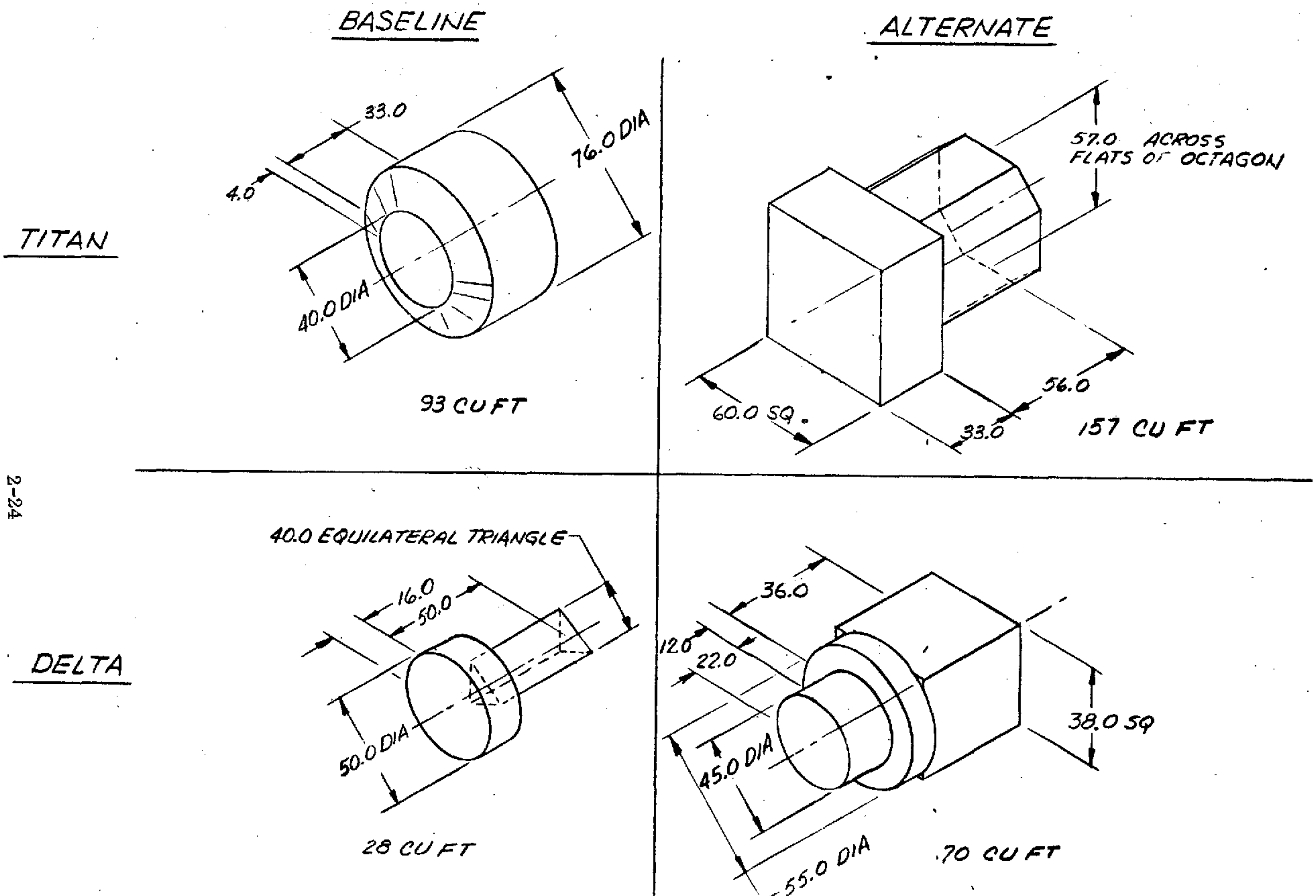

Figure 2-6. Propulsion System Space Allocations

Table 2-10. Evaluation Criteria

Shuttle Compatibility

- o Direct Shuttle access
- o Recover @ lower orbit to share launch cost

Mission Compatibility

- o Global coverage
- o Real time coverage of USA
- o Minimum sidelap (5-15%)
- o Repeat cycle (15-18 days)
- o Access time (2-4 days)
- o Minimum offset pointing ($\leq 30^{\circ}$)

Ground Station Compatibility

- o Minimum Orbit Adjusts
- o Need for ground station additions

Launch System Impacts

- o Transportation costs
- o System flexibility
- o Ease of transition to Shuttle
- o Launch system performance

Spacecraft Impacts

- o Altitude effects on spacecraft subsystems
- o Impacts on spacecraft as a function of launch vehicle

Instrument Impacts

- o Altitude effects on instruments
- o Impacts on instruments as a function of launch vehicle

Impacts of Later Missions

2.6.1 SHUTTLE COMPATIBILITY

The EOS-A spacecraft and orbit must be compatible with Shuttle retrieval even though the decision may be made not to retrieve the EOS-A spacecraft. The weight restrictions on an expendable launch vehicle or the long time lag between the EOS-A launch and the initiation of Shuttle operational capability from WTR will be considered when evaluating the advisability of retrieving EOS-A. Shuttle retrieval capability can be achieved in a number of ways:

- The orbit can be selected for direct Shuttle access thus limiting orbits to less than 450 nm. The allowable retrieval altitude is a function of the spacecraft retrieval weight.
- The orbit can be allowed to seek its desired altitude for other purposes and the on-board propulsion system sized to return the spacecraft to a Shuttle compatible orbit.
- The orbit can be selected for direct Shuttle access and the on-board propulsion system used to lower the Shuttle retrieval orbit to allow Shuttle payload sharing and therefore reduced Shuttle charges to EOS.

2.6.2 MISSION COMPATIBILITY

There are a range of desired mission orbit parameters that are discussed in detail in Section 3.0. The parameters that have the most significant impact on the launch vehicle and orbit selection are:

Global Coverage. This requirement dictates that the spacecraft must be able to achieve global data by the use of TDRS, tape recorders or local readout and defines the sidelap requirement at the equator.

Real Time Coverage of USA. Real time coverage of USA provides a tradeoff of choice of ground stations, number of ground stations and orbit altitude.

Minimum Sidelap. The minimum sidelap specified is 5-15 % at the equator. The range specified is bounded at the lower limit by ACS and orbit control and at the upper limit by limiting additional processing load on the ground system.

Repeat Cycle. The repeat cycle range of 15 to 18 days provides the opportunity to gather data periodically under relatively similar conditions. This repeat cycle range greatly reduces the number of orbits acceptable for EOS-A.

Access Time. The access time requirement of between 2-4 days dictates the HRPI offset pointing angle since offset pointing provides the opportunity to achieve these reduced access times. As altitude decreases the access time increases for a given HRPI offset angle.

Minimum Offset Pointing. As stated above the HRPI offset pointing angle and access time are directly related. Image quality decreases as the HRPI offset pointing angle increases with a desired offset pointing angle set at no greater than 30°. Therefore, as orbit altitude decreases the access time must increase to maintain the HRPI offset pointing within desired limits.

2.6.3 GROUND STATION COMPATIBILITY

The choice of the number of ground stations required for real time coverage of the USA is directly related to orbit altitude. As altitude increases the number of ground stations required for real time coverage of USA decreases. At the ERTS altitude (500 nm) two stations (Goldstone and NTTF) just cover the continental USA. From 500 nm to 400 nm ground station substitutions are required to cover the continental USA while below 400 nm a third station is required. The need for spacecraft orbit adjusts increases as the altitude decreases. The operational complexity increases as orbit adjusts are required and it is desirable to keep orbit adjust frequencies to no more than once per month.

2.6.4 LAUNCH SYSTEM IMPACTS

The launch system, which includes the launch vehicle and on-board propulsion system, impacts the orbit selection by:

- o Placing restrictions on the allowable spacecraft weight as a function of orbit altitude,
- o Establishing the launch and retrieval transportation costs as a function of mission altitude and Shuttle retrieval altitude,
- o Placing restrictions on total system flexitility to select a wide range of instruments or spacecraft subsystem designs as impacted by the launch vehicle shroud restrictions.

- o Limiting the ease of transition to Shuttle retrieval or ultimate launch, and retrieval and/or service by Shuttle.

2.6.5 SPACECRAFT IMPACTS

Spacecraft subsystem costs and weights are impacted by the choice of mission altitude. The ACS and Power subsystems prefer higher altitudes while C&DH and Wideband prefer lower altitudes. The cost impact of variations within the subsystems must be factored into the mission altitude selection.

2.6.6 PAYLOAD INSTRUMENT IMPACTS

Parametric instrument costs and weights can be established as a function of mission altitude, however, any such ideal curve of cost and weight vs. altitude must be tempered with engineering judgement. In many cases the instrument designs will be constrained to a discrete number of aperture sizes to use existing equipment and provide minimum non-recurring instrument costs.

2.2.7 IMPACTS OF LATER MISSIONS

The choice of mission altitude and launch vehicle for EOS-A must be compatible with the requirements for missions beyond EOS-A if a cost effective program is to be established. Therefore, the choice for EOS-A must be evaluated as to its compatibility with these later missions.

SECTION 3.0

MISSION ORBIT ANALYSIS

3.1 INTRODUCTION

There are a multitude of orbits which potentially meet the EOS mission requirements. These range from the ERTS-type (18 day repeat cycle, daily progression of the ground track) which are adequate for use with the Thematic Mapper, to various types of interlaced orbits which can provide shorter access time to points on the Earth when using the HRPI instrument. The parameters of interest when selecting the EOS-A orbit are defined in Table 3-1 along with some discussion as to how each of these parameters affects the EOS-A mission and system. Nominal ranges of requirements for each of these parameters are also identified in the Table.

3.2 ANALYSIS

Figure 3-1 defines all of the possible repeating orbits in the altitude range of 100 to 650 nautical miles. Consistent with the parameters and requirements listed in Table 3-1, the range of orbit investigated was quickly limited to the small clear area shown in the central portion of the figure. This is because:

- o Below 300 nm, there is very high atmospheric drag causing both excessive torques on the spacecraft and the need for almost constant orbit maintenance.
- o Above 500 nm there are no repeating orbits whose similar characteristics cannot be found in the range of 300-500 nm. Since instrument optics size increases as a function of altitude, the higher altitudes were not further investigated.
- o Given a 100 nm swath width as specified for the system, many orbits in the range of 300-500 nm can be eliminated, since they do not have sidelapping coverage on a global basis. These are all the orbits in the lower portion of the figure.
- o Correspondingly, for a 100 nm swath width, orbits can be eliminated for having excessive sidelap. These are the orbits in the upper portion of the figure. Note that sidelap increases with length of the repeat cycle, and, since decreased repeat cycle time is a general objective, the same orbits are less desirable for this reason also.

All orbits within the range of consideration then, have the following characteristics:

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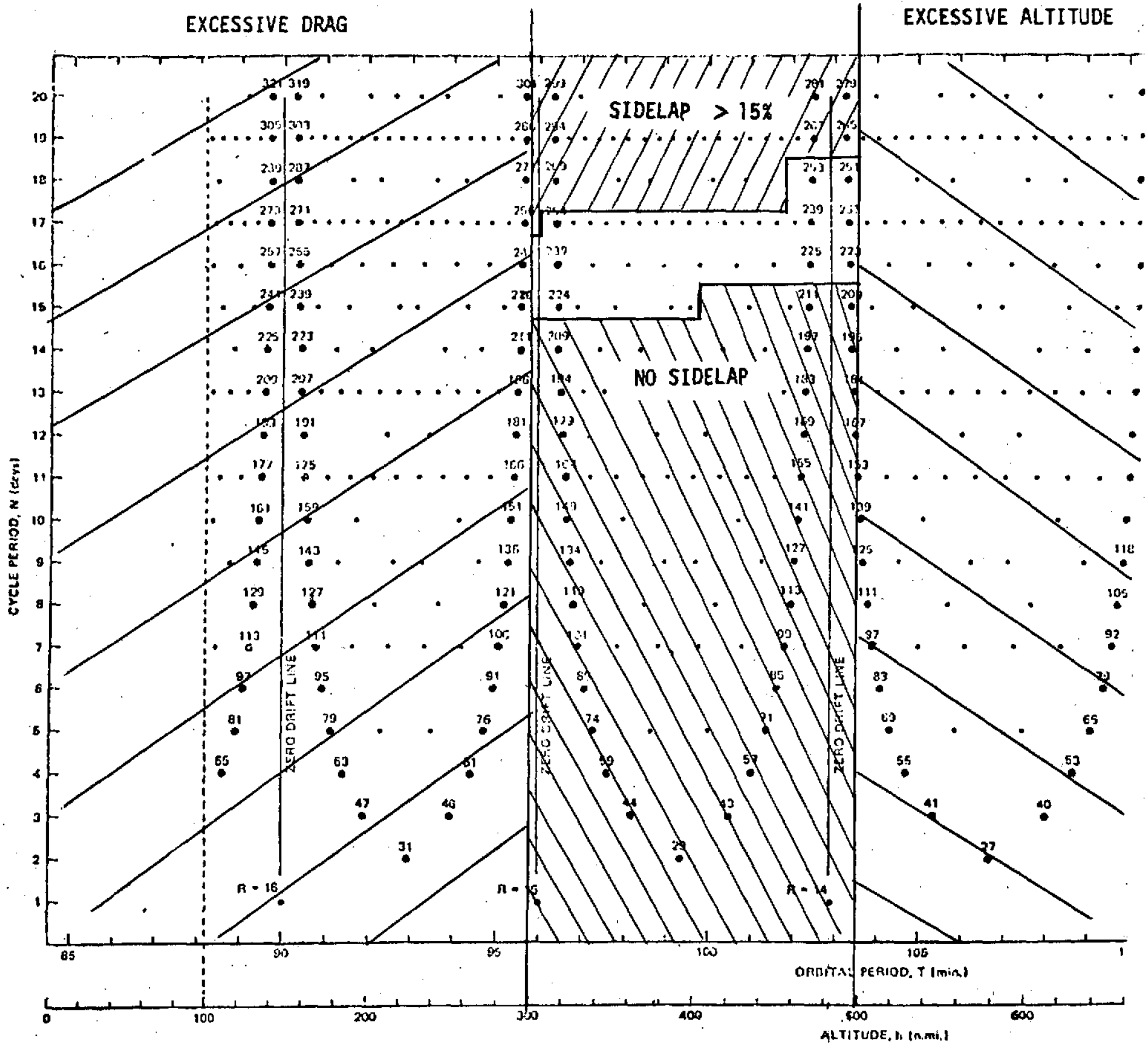
Table 3-1. Orbit Tradeoff Parameters

PARAMETERS	DEFINITION	SIGNIFICANCE FOR EOS-A	NOMINAL EOS REQUIREMENTS
Coverage	The amount of Earth which can be imaged by the satellite instruments.	Sun-synchronous orbits provide the opportunity for the satellite to view the entire earth up to approximately 81° latitude. Amount of coverage is then determined by the swath width.	Global
Swath Width	The width of sensor field of view as projected on the ground.	Determines the width of the imaged data and is a key trade-off parameter in sizing the instrument optics and detectors.	185 Km
Sidelap	The overlap between adjacent swaths as projected on the ground. Normally expressed as a percentage of the swath width and measured sidelap at the equator; sidelap increases with latitude.	Sufficient sidelap insures that no Earth surface area is lost between the swaths. ACS tolerance & orbit control precision will cause actual sidelap to vary about the nominal value. The more accurate the ACS & orbit control, the smaller the sidelap requirement. Excessive sidelap, while insuring coverage creates additional processing load on the ground system. ERTS had 14% sidelap. EOS ACS and orbit control performance can easily accommodate 5% sidelap.	5-15% at equator
Sun Synchronous	An orbit in which the orbital plane rotates at the same rate as the mean rate of the Earth about the sun.	Any point on the Earth will always be imaged at the same mean sun time which minimizes illumination changes on the ground scene. Also simplifies the power subsystem design by permitting one axis drive on the solar array & the thermal & ACS designs by limiting the variation in sun angle.	Sun Synchronous
Descending Node Time	The mean solar time the satellite crosses the equator on the North to South portion of its orbit. This terminology used primarily with sun synchronous orbits.	Affects Beta angle variation which in turn affects both thermal & power subsystem design. From user viewpoint, affects illumination conditions of the scene. Can be selected independently of all parameters.	09 30 - 12 30
Repeat Cycle	The time required for the satellite to (almost) exactly repeat its coverage pattern.	Provides the opportunity to gather data periodically under relatively similar conditions. Simplifies flight operations and data processing. Users generally prefer shorter repeat cycles.	15 - 18 days
Access Time	The minimum time between two sequential observations of the same point on the ground; it need not be under the same relative conditions. Sometimes defined as the time between an event occurence and when it can be observed.	Dictates HRPI offset pointing angle since offset pointing provides the opportunity for reduced access time. (Note in an ERTS-type orbit, repeat cycle equals access time.)	2 - 4 days
Interlace Pattern	The pattern in which successive ground tracks "fill in" between two adjacent orbits on a given day during one full repeat cycle.	Determines the time between the imaging of adjacent ground swaths. For a given access time this time can range from the access time up to one half the repeat cycle. The longer the time between adjacent swaths, the more the scene characteristics can change hence the poorer the mosaicing potential.	Equal to the access time
Altitude	The mean distance between the satellite orbit and the Earth radius.	Determines repeat cycle & interlace pattern. Affects instrument optics and detector sizing for a given coverage & swath width. Also affects launch vehicle payload capacity integral propulsion requirements and Shuttle service/recovery capability.	300 - 600 nm



R = number of revs per cycle
• = repeating orbit
279 ← number of orbits/repeat cycle
● = repeating minimum drift orbit

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FIGURE 3-1 CANDIDATE REPEATING ORBITS

(Extracted from "Swathing Patterns of Earth Sensing Satellites and Their Control by Orbit Selection and Modification," J. C. King Paper presented at Astrodynamics Specialists Conference - August 17-19, 1971)

- o Global coverage
- o 100 nm swath width
- o Sun synchronous
- o Descending node time - selectable
- o Repeat cycles between 15 and 18 days

The key tradeoff parameter to consider is altitude which in turn controls:

- o Drag
- o Direct Shuttle access
- o Interlace pattern and access time
- o Sidelap
- o U.S. Coverage in realtime

Considering the above parameters as criteria, the clear area of Figure 3-1 has been enlarged and divided into two portions, a "preferred" (clear) area and a "constrained" (shaded) area as shown in Figure 3-2. The preferred area was derived in the following manner:

- o Sidelap - areas with sidelap greater than 10% or less than 5% have been shaded in the upper and lower portion of the figure. Of all the tradeoff criteria to follow, sidelap in the range of 10-15% is by far the softest, i.e., if an orbit would otherwise be acceptable but have sidelap in this range, the constraint could easily be relaxed. The cost impact is increased data to be processed in the central data processing facility.

- o Drag - drag influences primarily how often orbit adjustments must be made. It is desirable to minimize the number of adjustments to simplify OCC operational support, GSFC computational loading and network support. In addition, the longer the spacecraft can operate without need for an orbit adjust, the more "failsafe" the orbit is. It's important to note that this "failsafe" orbit became a prime objective of orbit development on the ERTS program as the utility of long term repeating coverage became recognized. The figure has an arrow which shows the altitudes below which more than 1 orbit adjust per 16-17 day repeat cycle must be made. A similar arrow is shown for the altitude below which an orbit adjust must be made every two repeat cycles. At 500 nm altitude, the spacecraft can operate approximately 0.5 year without an orbit adjust (this is the "failsafe" orbit).

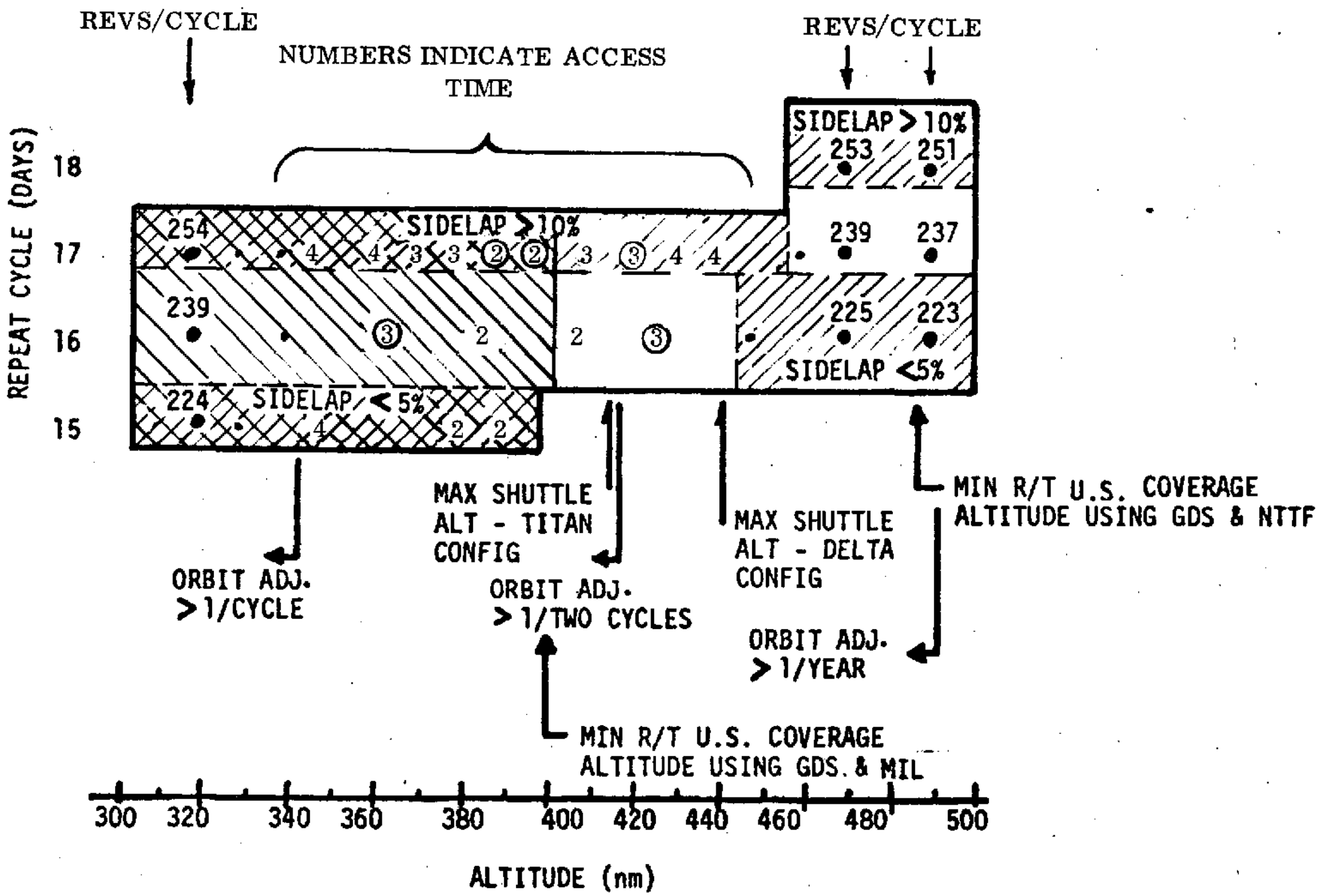

Figure 3-2. Candidate EOS-A Orbit

- o Access time/interlace pattern - the numbers in the central portion of the chart represent the access time for those particular orbits assuming use of a HRPI instrument with sufficient offset pointing capability. All two day access orbits are regular patterns with minimum time between adjacent swaths (equal to the access time). The three day access orbits are all regular patterns also (time between adjacent swath equal to three days) with the exception of the inner two on the 17 day repeat line. These both have time between adjacent swaths of five days. The inner four day access 17 day repeat orbits are also not regular, having 7 days between adjacent orbits. The other four day access orbits are regular. In general, the regular patterns with minimum time between adjacent orbits are preferred.
- o Continental U. S. Coverage - this is a very important operational constraint. When using the NTTF and Goldstone stations to receive realtime payload data, the edge of coverage occurs along the Gulf coast with its actual position a function of the orbit altitude. The edge of coverage occurs <u>within</u> the U. S. at altitudes below approximately 490 mautical miles. This becomes a major constraint in any "US in realtime" system. Figure 3-2 shows this constraint at the actual edge of coverage. When considering the normal operating procedures of a ground station, i.e., terminate transmission sometime prior to actual LOS to reconfigure the spacecraft (the number used on ERTS is 40 seconds and indeed creates problems along the coast of Louisiana), the constraint is even tighter. An orbit altitude at 490 nm implies wideband transmission right up to the edge of the cone (LOS). Orbits below 490 nm altitude cannot acquire all U. S. data in realtime using the Goldstone and NTTF stations. At 300 nm altitude, for example, portions of Texas, Louisiana, Mississippi, Arkansas and Oklahoma will not be accessable in realtime.

This last constraint means that

1) The only acceptable orbits are at or above 490 nm altitude, or
2) Some other solution must be found to solve the "U.S. in realtime" problem. Some of these other solutions include:
   a. Use TDRSS
   b. Use a WBVTR to record those portions of the U.S. not accessable in realtime.
   c. Add a fourth ground station or switch to a station different than NTTF or Goldstone to improve the coverage.

The first possible solution results in no orbit altitude constraints on the spacecraft. The second can solve the problem but creates real operational problems in the use of the tape recorders. The third alternate imposes some restrictions. The only candidate station is Merritt Island, Florida. (Corpus Christi would have been a candidate also, but it is being phased out of the STDN). Merritt Island can fill in the gap at altitudes above 300 nm and could replace the NTTF station at altitudes above 400 nm. This later case has the advantage that only three stations (Alaska, GDS and Merritt Island) must be equipped with wideband receive/record capability rather than four. It has the disadvantage that it imposes an effective 400 nm lower limit on satellite altitude. These "U.S. Realtime Coverage" constraints are shown on Figure 3-2 also.

The clear areas remaining on the figure contain those orbits which are optimum from a purely mission operation sense. They can be catagorized as:

- o Preferred interlaced 16 day repeats
- o Preferred non-interlaced 17 day repeats

There are also several slightly less preferred interlaced 17 day repeats with sidelap greater than 10%. The characteristics of these orbits are summarized in Table 3-2. The final selectior of altitude can now be made considering:

- o Shuttle compatability
- o Instruments considering the need for and quality of offset pointing data.
- o Some subjectivity when things are equal (or marginal).

Figure 3-2 shows the maximum direct Shuttle access altitude that can be attained using the weigh of both Delta and Titan configured all-up (TM and HRPI) spacecraft. These altitudes are 440 nm and 415 nm respectively. Any orbit below these altitudes is directly accessable by Shuttle for both launch and retrieve. The lower the selected altitude, the greater the weight margin available for the given configuration.

In addition to the all-up Shuttle launch/retrievable Delta and Titan spacecraft, one other configuration was considered, i.e., a TM only non-retrievable mission. The recommended orbits for these three cases are described in paragraph 3.3.

Table 3-2. Parameters for Candidate Orbits

Nominal Access Time (days)	Days To Repeat	Interlace Factor	No. Revs. Per Cycle	Revs. Per Day	Nodal Separation	% Sidelap For 100 nm Swath	Altitude nms	Orbital Period (sec)	Minimum Time Between Adjacent Swath (days)	Offset Pointing Angle (°)
2	16	7	231	14.4375	24.9351	6.5	403	5984	7	±45
3	16	5	229	14.3125	25.1528	5.7	425	6037	3	±32
3	17	6	244	14.3529	25.082	11.5	418	6020	3	±32
3	17	7	245	14.4118	24.980	11.8	407	5995	5	±32
4	17	4	242	14.2353	25.2892	10.8	439	6069	4	±27
4	17	5	243	14.2941	25.1852	11.1	428	6044	7	±27
17	17	-	239	14.0588	25.6067	10	473	6146	1	-
17	17	-	237	13.9412	25.8230	9	494	6197	1	-

3.3 RECOMMENDED ORBITS

TM Only, No Shuttle Launch/Retrieve. This configuration was considered in the study in order to estimate the costs for a non Shuttle launch/retrieve spacecraft. Without use of the Shuttle, the orbit is no longer constrained to the lower altitudes, i.e., altitudes in the 500 nm regime are reasonable for payload weights in the range of 2000 to 4000 lbs. In addition, since the HRPI is not part of the payload, there is no need for interlaced orbits to support offset pointing. Hence, the preferred orbit characteristics include an ERTS-type swathing pattern (adjacent swaths on successive days) and the minimization of drag. There are two orbits shown on Figure 3-2 which have these characteristics. They are at 473 and 494 nm altitudes respectively. The only difference between the two is that the lower orbit has an easterly progression while the 494 nm orbit has a westerdly progression. Westwardly progression is advantageous in that the orbit pattern moves in an opposite direction to major weather patterns and hence is less likely to track or move with cloud masses. In addition, the 494 nm orbit is the only one which can provide coverage of the entire continental U.S. in realtime with the NTTF and GDS stations. Hence, 494 nm orbit is preferred for the TM only, no Shuttle launch/retrieve case. Its swathing pattern is shown in Figure 3-3.

Delta Configuration Direct Shuttle Retrieve. Figure 3-2 shows that any orbit below 440 nm is directly Shuttle accessable for an EOS-A Delta configuration spacecraft. It's important, however, to recognize the steepness of the Shuttle weight/altitude performance curve; small changes in Shuttle payload capability reflect themselves in significant changes in altitude. The Shuttle payload launch capability shown in Figure 5-7 Section 5.0 was used to derive the maximum altitude points shown in Figure 3-2.

The inclusion of the HRPI in the payload demands an interlaced orbit to provide minimum access time. Two, three and four day access times were investigated to determine image quality as a function of offset pointing angle. The offset pointing angles of the two, three and four day access orbits are approximately 27, 32, and 45 degrees respectively. Investigation of image quality as a function of offset pointing angle demonstrated (Report 2 Section 3.4.2) that quality degrades rapidly becoming unusable at offset pointing angles beyond 35 degrees. In terms of useful imagery then, two day access buys very little and the optimum access time is 3 days. (Note that the 3 day access time improves to two days at the higher latitudes. The preferred orbits then are the 16 day repeat, 3 day access, 425 nm orbit, and the 17 day repeat, 3 day access, 418 nm orbit. The former has 5.7% sidelap, the latter has 11.5%. The choice between them is a tossup. The latter more closely fits the generally acceptaed 10% sidelap value with the former somewhat reducing the load on the ground station. The 418 nm orbit was used as the baseline throughout the later phase of the EOS study and its swathing pattern is shown in Figure 3-4. The 425 nm orbit has a very slight advantage providing the altitude control requirements are not substantially changed from the .01 degree pointing accuracy.

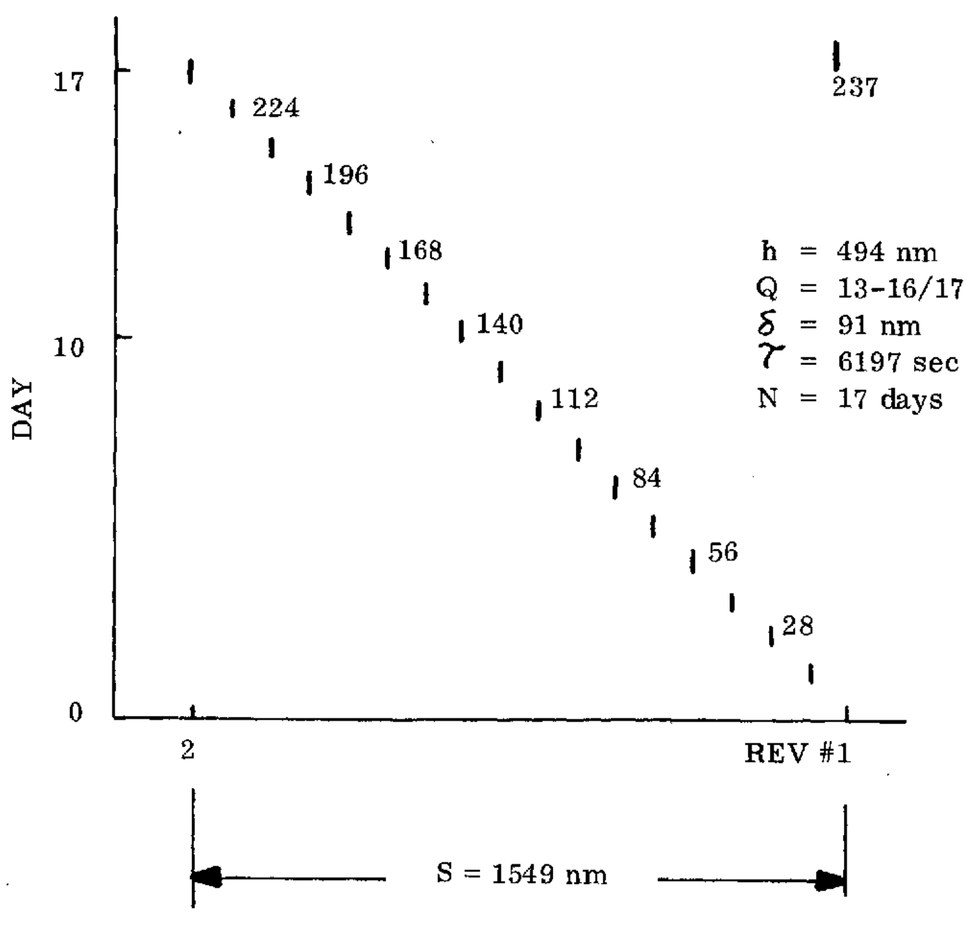

Figure 3-3. Preferred TM (Only) Orbit (No Shuttle Launch/Retrieve)

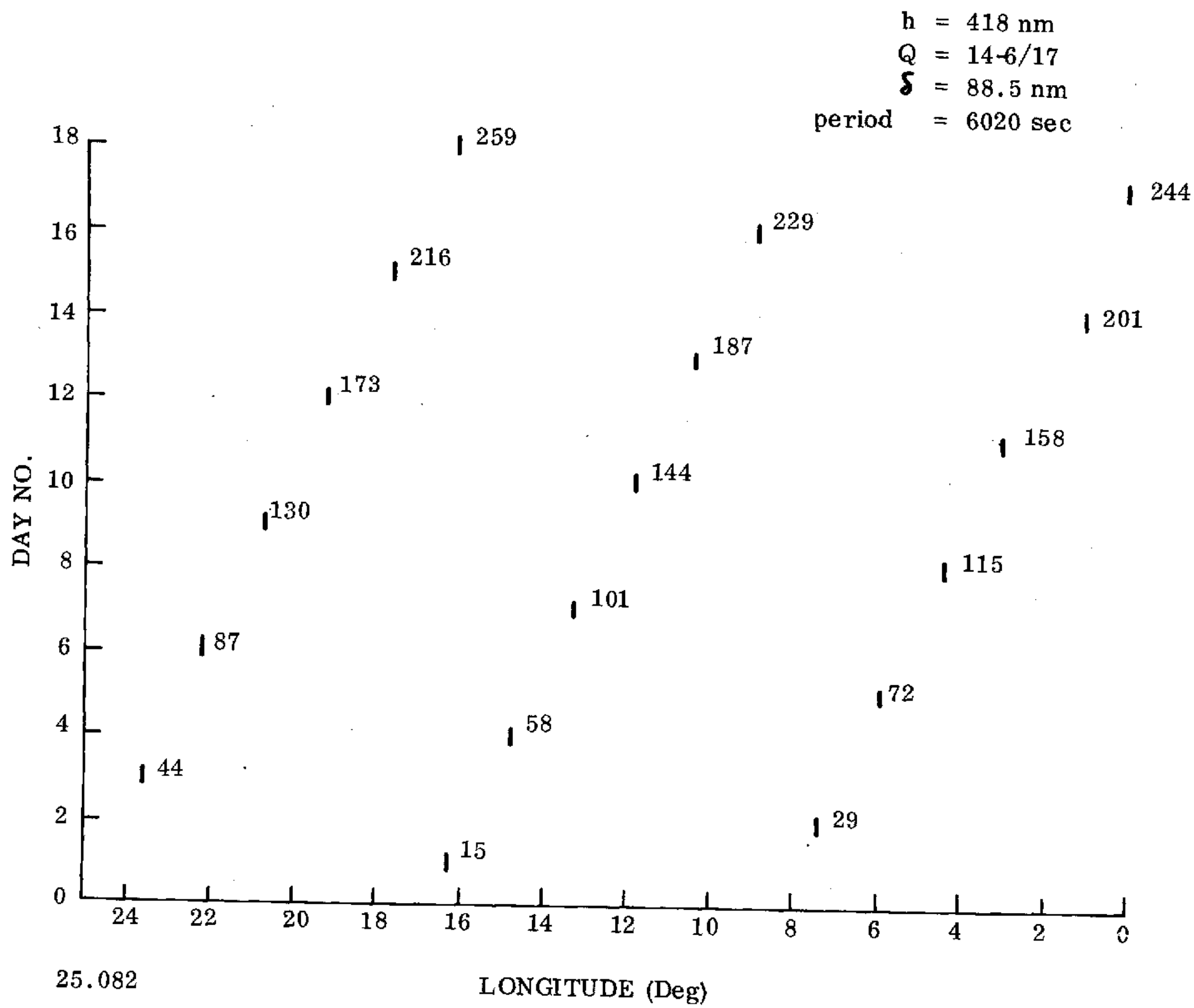

Figure 3-4. Three Day Access Orbit

Titan Configuration - Direct Shuttle Retrieve. Figure 3-2 shows that any orbit below 415 nm is directly Shuttle accessable for an EOS-A Titan configuration spacecraft. Again the payload includes a HRPI and a 3 day access orbit is required to take maximum advantage of the offset pointing capability. Figure 3-2 shows only one 3-day access orbit available, the 17 day repeat at 407 nm altitude. This orbit is somewhat less than optimum in that it is not regular - there are 5 days, not 3, between adjacent orbits. There is also very little apparent weight margin. The wieght margin problem is not so bad as it appears from the chart, however. The Shuttle launch weights were derived assuming a simultaneous launch and retrieve of EOS type spacecraft (a spacecraft launched by the same Shuttle which retrieves the spacecraft under consideration here). If an empty Shuttle is launched, or one using less than full capacity, the maximum altitude for retrieve increases with the exact amount depending on the percentage of Shuttle capacity used at launch. In fact, it is very probable that sufficient weight margin can be planned to enable an EOS-A retrieval from a 418 or 425 nautical mile orbit, which were shown to be the optimal orbits for the mission. Hence, the recommended EOS-A orbit for an all-up spacecraft, either Titan or Delta launched and Shuttle retrieved, is 418 nm.

SECTION 4.0

ORBIT ALTITUDE EFFECTS ON SPACECRAFT AND PAYLOAD PERFORMANCE AND COST

4.1 INTRODUCTION

Orbit altitude effects on spacecraft and payload design has been evaluated over an altitude range of 280 to 900 nm.

Specific spacecraft performance/cost trades were done for:

- o Power Subsystem
- o ACS Subsystem
- o Wideband/C&DH Subsystem
- o Thermal Subsystem
- o Propulsion Subsystem

Payload performance trades were done using the Thematic Mapper as a representative instrument. Instrument/altitude cost trades were done on a relative basis.

4.2 SUMMARY AND CONCLUSION

The following summarizes the cost trades for the various subsystems. Section 4.3 contains the analysis for these trades. It is significant that no "driving" cost factors were identified for any of the spacecraft subsystems that would constrain orbit altitude selection.

4.2.1 POWER

Power subsystem performance is impacted by altitude changes because of the effects of both particle radiation damage and orbital period/dark time upon the solar array. End of two year performance can be kept constant with altitude by selecting a specific sized array for each altitude. Performance analysis indicated that an optimum altitude exists which results in the minimum sized solar array and hence lowest cost (≈ 580 nm). Any deviation from this "optimum" altitude carries with it a cost penalty; however, this penalty is small ($10K) across the range of altitude considered as shown in Table 4-1.

Table 4-1

Altitude	Cost of Array
580 nm	0
350 nm	+4%
850 nm	+4%

4.2.2 ACS

At 915 nm, the altitude pointing accuracy degrades by approximately 0.6 arc seconds from the nominal value at 418 nm. This error is insignificant compared to the system pointing requirement of 36 arc seconds.

At altitudes lower than 350 nm, the increase in aerodynamic drag requires the momentum wheels to handle an additional 2.0 lbs/ft/sec of sinusoidal angular momentum and the magnetic unloading system to unload an additional 0.27 lbs/ft/sec of angular momentum every hours.

The sinusoidal momentum content can be accommodated with the standard ACS, but an additional 20,000 pole-cm capability must be added to the pitch axis magnetic torquer. The weight increase is approximately 6 lbs. This change results in little cost impact.

4.2.3 WIDEBAND/C&DH

Wideband/C&DH parameters such as antenna gain and transmitter power can be cost/performance traded with altitude for lowest cost. Since antenna cost is smaller than power amplifier cost, it is most economical to use the largest size antenna which can be packaged within the available envelope, thereby using the lowest power transmitter for any given EIRP. This relationship holds until the antenna size increases to the point that highly accurate pointing and control are required. At this point antenna cost increases more rapidly than more powerful transmitters. Total system costs, however, are highly dependent on the availability of developed hardware, and totally parametric trades can be misleading. For several discrete cases analyzed, system recurring costs increased about $100K over the altitude range considered; minimum cost, of course, occurring at the lowest altitude.

4.2.4 THERMAL

Thermal subsystem performance changes with altitude because of increased or decreased effects of earth albedo on portions of the spacecraft that view the earth. Increased altitude results in decreased albedo and hence a cooler spacecraft. The magnitude of thermal change between 280 to 900 nm is not large and can easily be compensated for in the spacecraft design by increasing or decreasing the spacecraft thermal radiator surface area depending on altitude. This radiation surface area change is so small that there is no cost or weight impact with altitude changes.

4.2.5 PROPULSION

Propulsion system design is significantly impacted by the orbit altitude. The major variables to be considered are the orbit adjust fuel requirements which increase as drag becomes more significant at lower altitudes and the orbit transfer requirements which increase at the higher altitudes assuming that the spacecraft is required to return to Shuttle at approximately 300 nm. There are large changes in propulsion system weights as the mission altitude and Shuttle altitude are varied. These are discussed in detail in the Launch System section - Section 5.0. The propulsion system cost is relatively insensitive to the total Δ V requirements and therefore insensitive to the mission altitude variations. If orbit transfer is not required, the system can be simplified eliminating the large orbit transfer engine giving a slight cost reduction as illustrated in Table 4-2.

Table 4-2

Altitude	Cost of Propulsion (Shuttle Retrieve at 300 nm)	Cost of Propulsion (Shuttle Retrieve at Mission Altitude)
300 nm	-20%	None
400 nm	Reference	Reference
500 nm	None	Shuttle Retrieval not Possible

4.2.6 PAYLOAD

Table 4-3

Altitude	Cost of Instrument
270 nm	- 18%
418 nm	Reference
540 nm	+ 22%

Payload cost is sensitive to orbit altitude changes when the standard performance parameters of SNR, ground resolution and swath width are maintained constant. Payload design factors impacted by altitude changes are the number of detectors, aperture size and focal length. Estimates of the cost sensitivity are highly variable. The highest cost sensitivity to altitude for idealized designs are illustrated in Table 4-3. In practice, however, the instrument designs will be constrained to a discrete number of aperture sizes to use existing equipment. This greatly reduces the variation of cost with altitude over the altitude range of 350 to 420 nm.

4.3 PERFORMANCE ANALYSIS

4.3.1 ORBIT ALTITUDE EFFECTS ON THE POWER SUBSYSTEM

The power subsystem is affected by two major factors which are a function of orbit altitude: (1) particle radiation damage, and (2) orbit period and dark time. The dependence of the latter factors on orbit altitude is shown in Figure 4-1. The increase in the ratio of sunlight time-to-dark time as the orbit altitude increases has the effect of reducing the required solar array area for a given load power demand. The particle radiation environment consists of a solar flare proton component and a trapped particle component which is a function of orbit altitude. The solar flare proton integral spectra is given in Figure 4-2 for a single anomalously large (AL) event. It was assumed that one such event occurs during the two year design lifetime. The free space model for this AL event was reduced as shown on the figure to account for the shielding afforded by the earth's magnetic fields. The trapped particle radiation environment was obtained from the applicable volumes of NASA SP-3024 for the range of altitudes of interest. These particle environments were translated into solar cell damage equivalent 1-MeV electron fluences. The resulting damage equivalent 1-MeV electron fluences are shown in Figures 4-3 and -4 for the trapped

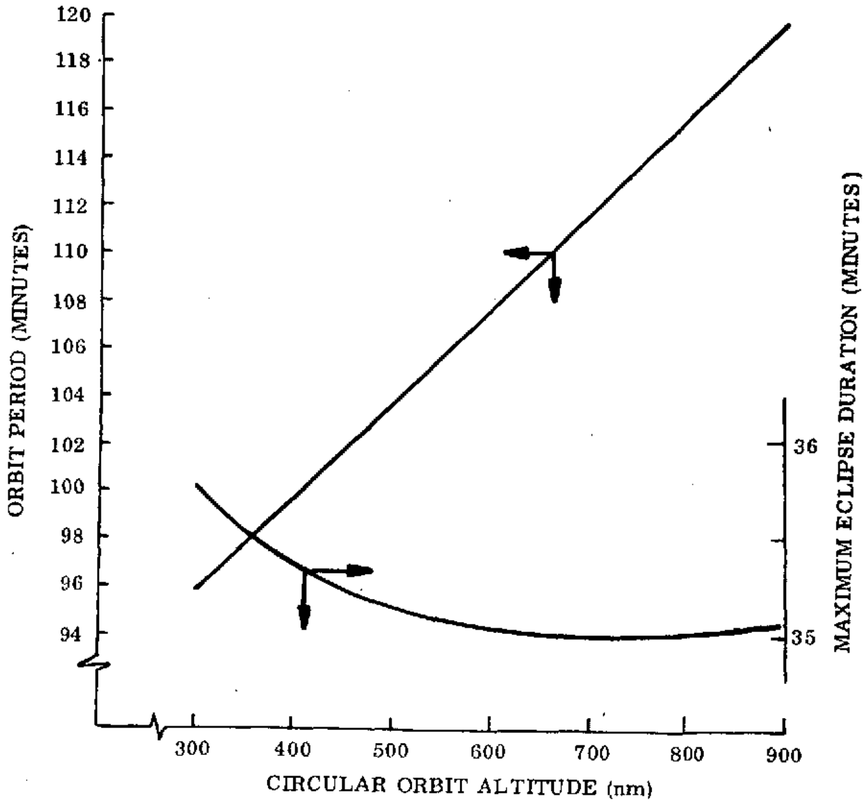

Figure 4-1. Orbit Period and Maximum Eclipse Duration vs. Circular Orbit Altitude

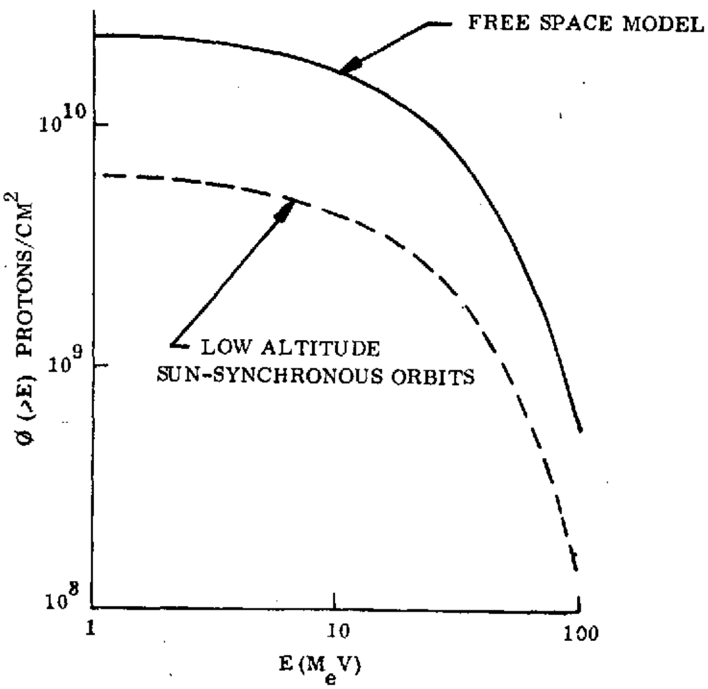

Figure 4-2. Solar Flare Proton Integral Spectra for a Single Anomalously Large Event

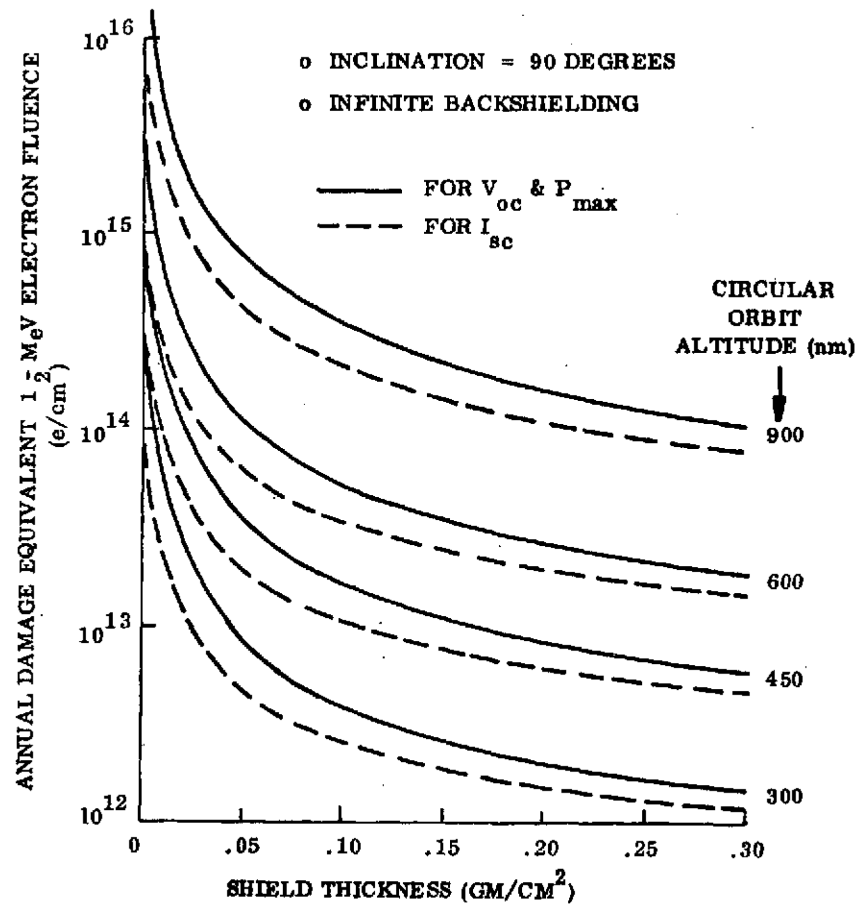

Figure 4-3. Annual Damage Equivalent 1-MeV Electron Fluence for Trapped Particles vs. Shield Thickness for Several Orbit Altitudes

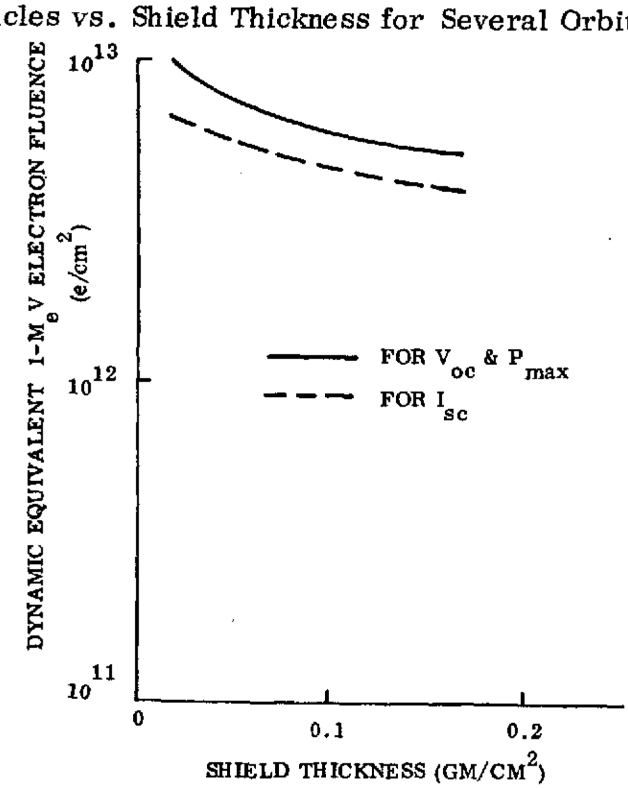

Figure 4-4. Damage Equivalent 1-MeV Electron Fluence for a Single Anomalously Large Solar Flare Event in a 300 to 900 nm Sun-Synchronous Orbit

particle and solar flare proton environments, respectively. These damage equivalent 1-MeV electron fluences were translated into solar cell open-circuit voltage (V_{oc}), short-circuit current (I_{sc}), and maximum power (P_{max}) degradations as shown in Figures 4-5 4-6, for 6 and 12 mils of fused silica coverglass thickness, respectively. In both cases the back shield was maintained constant at 15 mils of equivalent aluminum.

An analysis of the power subsystem performance yielded the results given in Figure 4-7 in terms of the solar array panel area required as a function of orbit altitude. A direct energy transfer (DET) power subsystem implementation was assumed with a load power profile which reflects EOS-A demand as shown in Figure 4-8. The duration of the peak loads was maintained constant at the values indicated as the orbit altitude was varied. As the results indicate, there is an orbit altitude which yields the minimum solar array panel area for a given shielding. The low end of the altitude range has relatively low radiation damage, but a higher ratio of dark time-to-orbit period. The higher altitudes have relatively high radiation damage, but a lower ratio of dark time-to-orbit period. These two major influences interact to produce the minimum solar array area as shown in Figure 4-7.

4.3.2 ORBIT ALTITUDE EFFECTS ON THE ATTITUDE CONTROL SUBSYSTEM

Orbit altitude impacts the ACS performance in two areas - star sensor update intervals, and disturbance torque momentum accumulation. The star sensor update interval affects the accuracy of the ACS, and the disturbance torque affects momentum wheel sizing and unloading requirements.

4.3.2.1 Star Sensor Update Interval

The EOS Attitude Control System uses a three-axis gyro package (inertial reference unit), star sensors, and the On-Board Computer (OBC) for attitude reference and control. The inertial reference unit is the primary source of attitude information to the OBC, but it provides only relative attitude information (i.e., attitude change). Absolute attitude information is provided by the star sensors which periodically transit known stars, and provide the OBC with star sighting errors. The OBC then converts these errors to spacecraft attitude errors and makes the appropriate corrections. The errors which accumulate between star transits are due primarily to inaccuracies in the inertial reference unit.

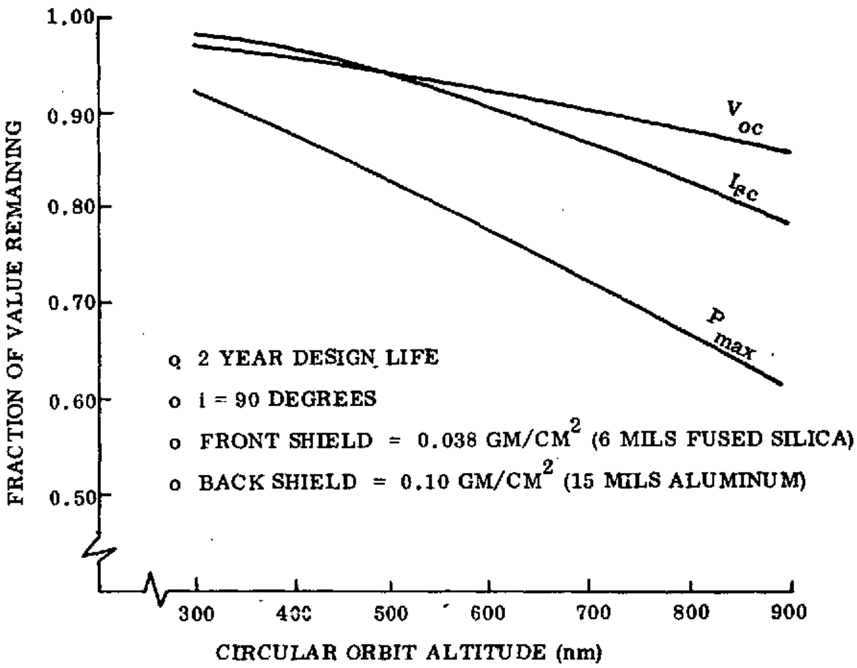

Figure 4-5. Solar Cell Degradation vs. Circular Orbit Altitude for 6 Mil Thick Fused Silica Coverglass

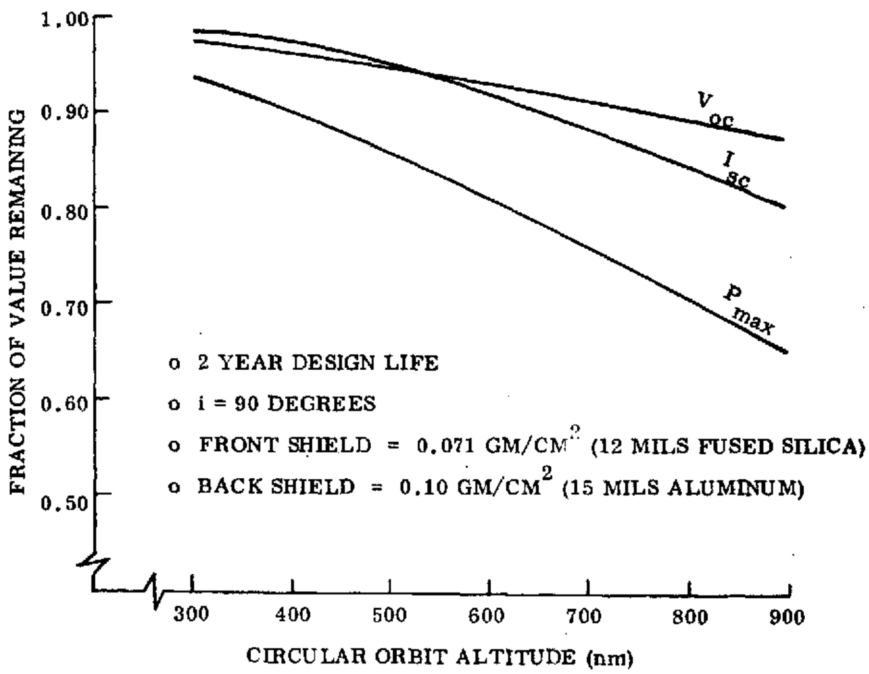

Figure 4-6. Solar Cell Degradation vs. Circular Orbit Altitude for 12 Mil Thick Fused Silica Coverglass

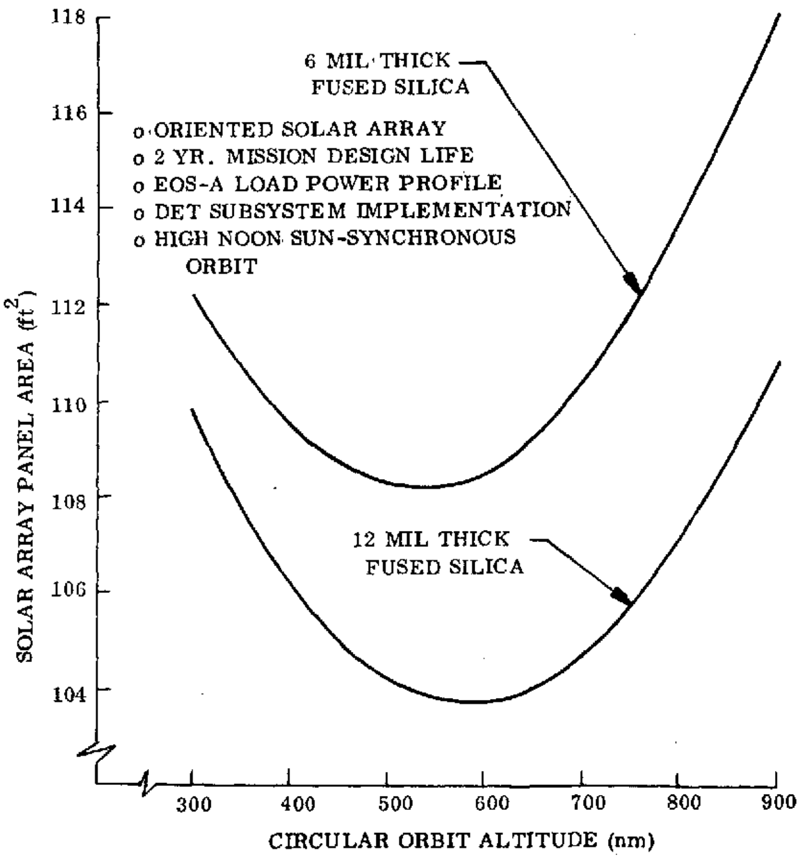

Figure 4-7. Solar Array Panel Area Required for EOS-A vs. Orbit Altitude

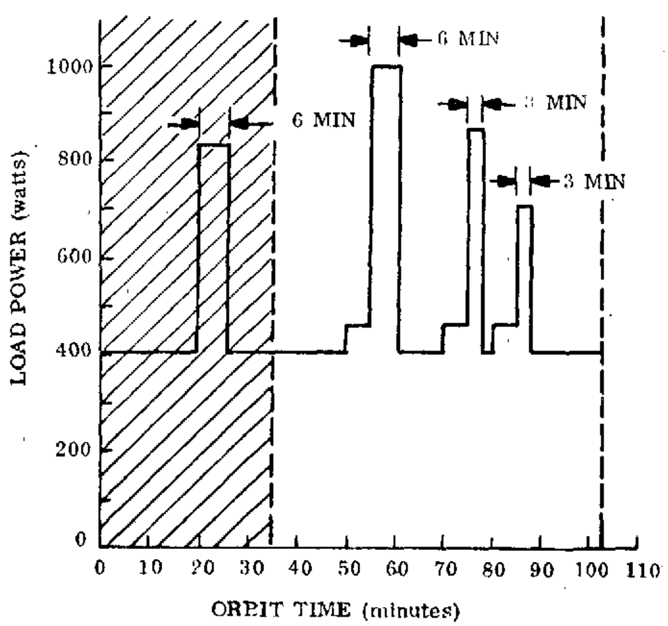

Figure 4-8. Typical Load Power Profile (EOS-A)

The limit on attitude accuracy is determined by the drift rate of the gyros, and the time between star transits. For the nominal (418 nm) orbit the standard time between updates has been selected as 1,000 seconds. As altitude increases, the time between star transits increases, permitting a longer period for gyro drift and hence lower system altitude accuracy.

For the orbits under consideration, the orbital rate extends from a maximum of .001096 rad/sec to a minimum of .000872 rad/sec. Hence the "standard" star update interval extends from 954 sec. to 1200 sec. The minimum orbital rate is the most important value, and with a nominal gyro drift rate (per specification) of 0.003 deg/hr, the peak attitude error will be 3.6 arc sec, representing an increase of approximately 0.6 arc sec over the nominal orbit.

4.3.2.2 Disturbance Torque Variation

Spacecraft are acted upon by external disturbance torques. These torques integrate to cause a momentum build-up which must be accumulated by the momentum wheels. Torques with components which are constant in inertial space (secular torques) integrate without bound, and the momentum wheels must be unloaded to maintain spacecraft control. Hence both the momentum wheels and the momentum unloading systems are affected.

At low altitudes, four torques are of major significance, solar torque, aerodynamic torque, gravity gradient torque and magnetic torque.

4.3.2.2.1 Solar Torque

Solar torque is caused by the pressure of sunlight (approximately 10^{-7} lb/ft^2) creating a force on the spacecraft which does not pass through its center of mass. Solar torque is totally configuration dependent and varies with spacecraft area, sun angle, reflectivity characteristics, etc. As a consequence, solar torque, and its resulting momentum accumulation, must be evaluated for a specific configuration.

For EOS-A, the major source of solar torque is the solar array. The solar array is physically located on the pitch (y) axis, with its center of pressure a maximum of fifteen feet from the spacecraft center of mass, along the -y axis. The array is controlled to point to the sun, and the solar force is therefore constant, independent of orbital position (excluding eclipse). Since the center of pressure - center of mass relationship remains constant in inertial space, solar torque is secular (i.e., constant in inertial space). The solar torque momentum, therefore, increases linearly with time and for all practical purposes without bound. The solar torque momentum accumulated by EOS-A is in the orbit plane, and is approximately in the north-south direction. Since momentum is a vector, and fixed in inertial space, the roll-yaw momentum wheels must interchange their momentum content as the spacecraft rotates in orbit. The result is the roll and yaw momentum wheels exhibit and orbital frequency sinusoidal characteristic with increasing amplitude, and 90° out of phase with each other.

Figure 4-9 shows the EOS-A roll momentum wheel history for three orbit altitudes: 300 nm, 418 nm and 915 nm. Each wheel illustrates the sinusoidal history mentioned earlier. Since solar pressure is independent of altitude, the inertial torque and momentum accumulation is independent of altitude. This is evident from Figure 4-9, since the momentum histories all have the same envelope irrespective of altitude. The envelope represents a momentum growth rate of approximately 0.6 lb-ft sec/hr. The yaw momentum wheel history,

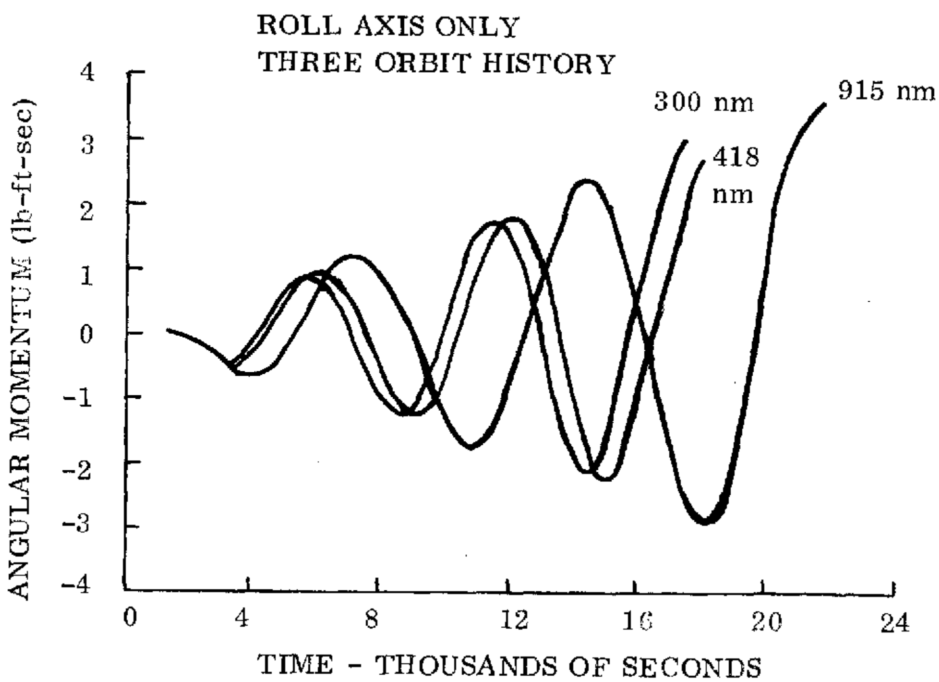

Figure 4-9. Momentum Accumulated from Solar Torque

which is not shown, is similar to the roll history, but out of phase. The pitch momentum wheel history, also not shown, is predominantly at orbital frequency (with a small secular component), and is less than 0.02 lb-ft-sec for any of the orbits. For the range of orbit altitudes being considered, therefore, solar torque effects do not change significantly and can be accommodated for by the ACS.

4.3.2.2.2 Aerodynamic Torque

Aerodynamic torque is caused by the aerodynamic pressure associated with the spacecraft's passage through the atmosphere. The torque results from the aerodynamic force (drag force) not passing through the spacecraft center of mass. Like solar torque, aerodynamic torque is totally configuration dependent, and must be evaluated for the configuration being considered. For EOS-A, the major source of aerodynamic torque is the solar array. The torque caused by the array is primarily about the yaw (z) axis of the spacecraft and is the same direction (on the z axis) at all points in the orbit. Consequently, the torque is not fixed in inertial space but rotates at orbital rate, and therefore produces a sinusoidal momentum.

Figure 4-10 shows the roll wheel momentum history for the 300 nm and 418 nm orbit. The sinusoidal characteristic of the momentum is evident, but the sine wave is heavily distorted. The distortion arises from the variation in the atmosphere density around the orbit. The density is highest during the day, and at its worst, is approximately five times more dense than the atmosphere at night. This density variation is also responsible for the secular momentum build-up which, from the envelope of the sine wave, is approximately 0.27 lb-ft-sec/hr for the 300 nm orbit.

The yaw wheel momentum history is not shown, but is similar to the roll wheel momentum history except for the phase difference. The pitch wheel momentum history is primarily secular but at 300 nm reaches only 0.1 lb-ft-sec after three orbits.

Two other characteristics of Figure 4-10 require explanation. First the momentum histories indicate a bias value of approximately -0.8 lb-ft-sec. The bias is the result of initial conditions selected for the simulation, and is not a "permanent" characteristic.

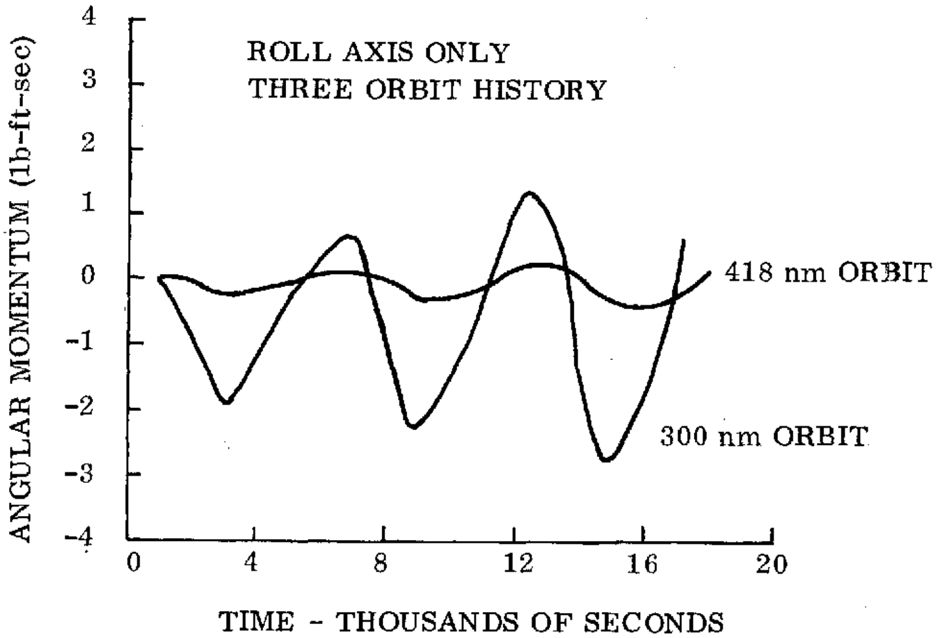

Figure 4-10. Momentum Accumulated from Aerodynamics

The second item is the rapid decrease in aerodynamic pressure with altitude. This is the most significant item from the analysis standpoint since it indicates that the 300 nm orbit imposes special requirements on theACS because of aerodynamic torque. First, the momentum wheels must be capable of accommodating an additional sinusoidal momentum content of approximately 2 lb-ft-sec, and the magnetic unloading subsystem must be capable of unloading an additional 0.27 lb-ft-sec/hr secular momentum growth. A preliminary estimate is that the momentum wheels for the 300 nm orbit will handle the sinusoidal momentum if the magnetic unloading system increases the pitch magnetic torquer capability to 50,000 pole-cm. The weight increases would be approximately six pounds.

4.3.2.2.3 Gravity Gradient Torque

Gravity gradient torques are caused by the gradient in the earth's gravitational field acting on the spacecrafts moments and products of inertia. For an earth oriented spacecraft, only two gravity gradient torques appear, one in the spacecraft pitch (y) axis, and one in the spacecraft roll (x) axis. There is never a torque about the local vertical, and hence no torque about the spacecraft yaw (z) axis.

The pitch gravity gradient torque is proportional to the spacecraft xz product of inertia, and since the pitch axis changes direction only slowly (in inertial space), the resulting pitch momentum increases linearly with time. The rate of growth decreases as the altitude increases because the gravity gradient torque is proportional to the square of the orbital rate (for a circular orbit). The pitch momentum accumulation for the three orbits under consideration is shown in Figure 4-11. The xz product of inertia for 25 slug-ft^2. With this value, the momentum accumulated in the 300 nm orbit is approximately 0.1 lb-ft-sec/hr higher than that accumulated in the 915 nm orbit.

The roll gravity gradient torque is a constant on the roll axis, and since the spacecraft is rotating at orbital rate, the roll torque vector rotates in inertial space. A steady state

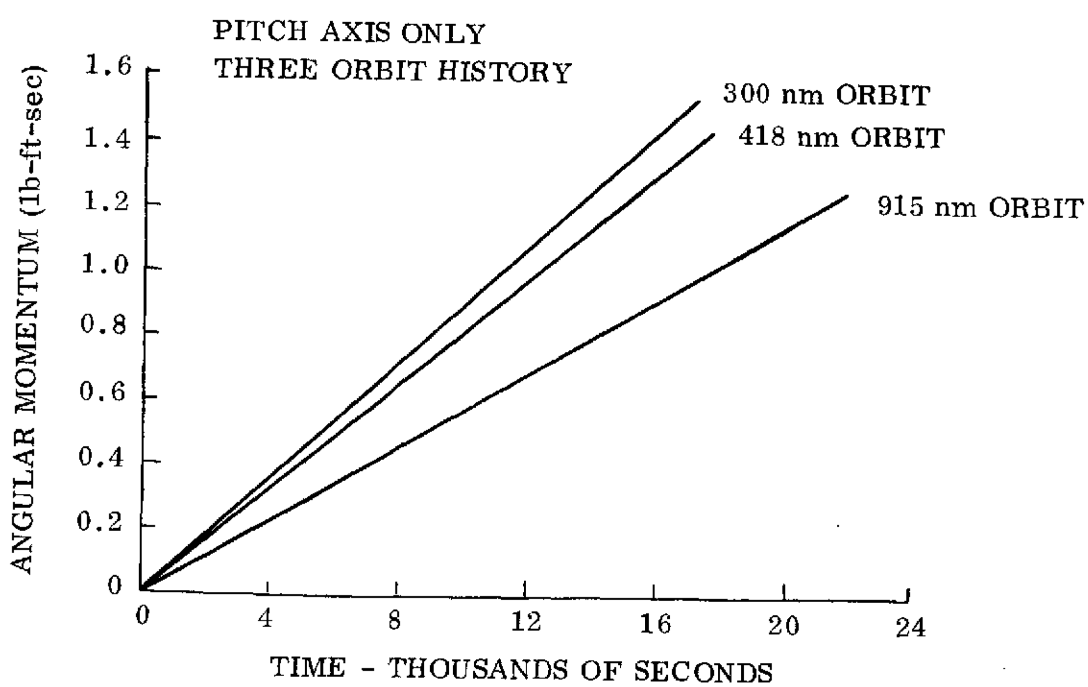

Figure 4-11. Momentum Accumulated From Gravity Gradient

dynamical condition is reached when the yaw momentum wheel reaches a constant momentum value. This value is proportional to the roll gravity gradient torque divided by orbital rate, and for the 300 nm orbit (worst case) is approximately 0.032 lb-ft-sec.

The change in gravity gradient momentum accumulation as a function of attitude is relatively small, and can be accommodated by the ACS.

4.3.2.2.4 Magnetic Torques

Magnetic torques are caused by a spacecraft magnetic dipole interacting with the earth's magnetic field. The magnetic torques are a function of the spacecraft dipole, orbital altitude, and inclination, and can be calculated directly once these values are known. In general, magnetic torques occur about all three spacecraft axis, and for an earth oriented spacecraft with a permanent magnetic dipole, all but one of the torques is sinusoidal.

For EOS-A, the spacecraft has been assumed to have a dipole equivalent to 10,000 pole-cm in each axis. The pitch wheel momentum history for this dipole is shown in Figure 4-12 and is obviously sinusoidal. There is a slight difference with altitude since magnetic torques like gravity gradient torques, are proportional to the square of the orbital rate. The torque is approximately 58% higher at 300 nm altitude than at 915 nm altitude. The sinusoidal content is only 28% higher, however, since it is directly proportional to the orbital rate.

The yaw wheel momentum history is shown in Figure 4-13. The curve exhibits the same increasing sinusoid as the solar torque curve, and for the same reason; there is a growth in angular momentum in the orbit plane. This accumulation is the result of the pitch axis dipole, of which approximately half is effective in causing a secular momentum growth. The growth in the 300 nm orbit is approximately 58% higher than that in the 915 nm orbit.

For a non-constant spacecraft magnetic dipole, the angular momentum accumulation is likely to be secular. A limit of 100 pole-cm has been placed on the non-constant portion of the spacecraft dipole to limit the momentum accumulation.

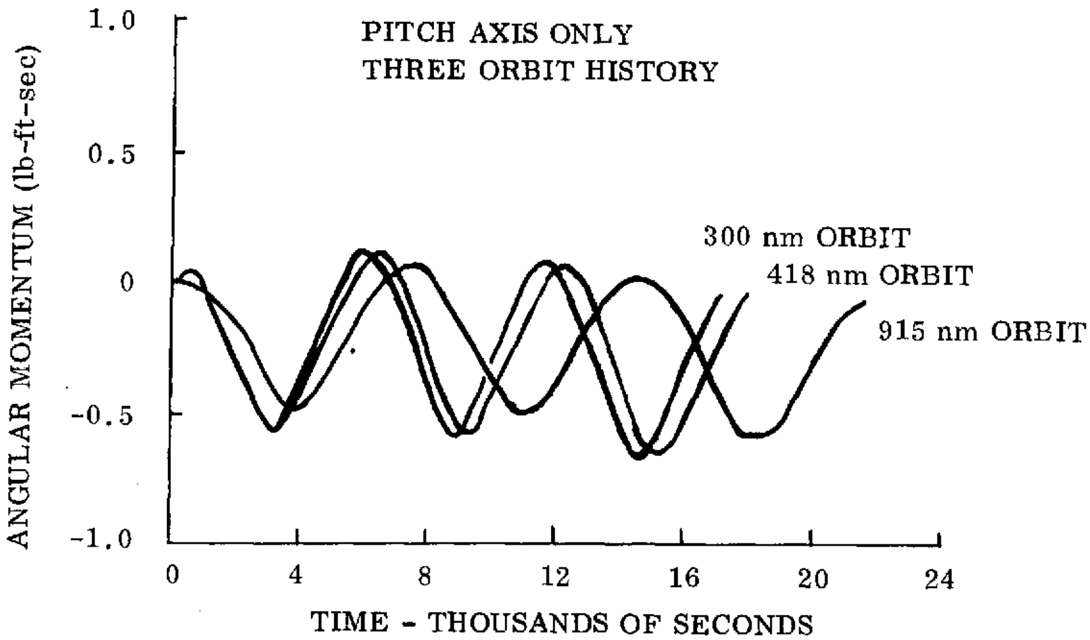

Figure 4-12. Momentum Accumulated From Magnetic Torque

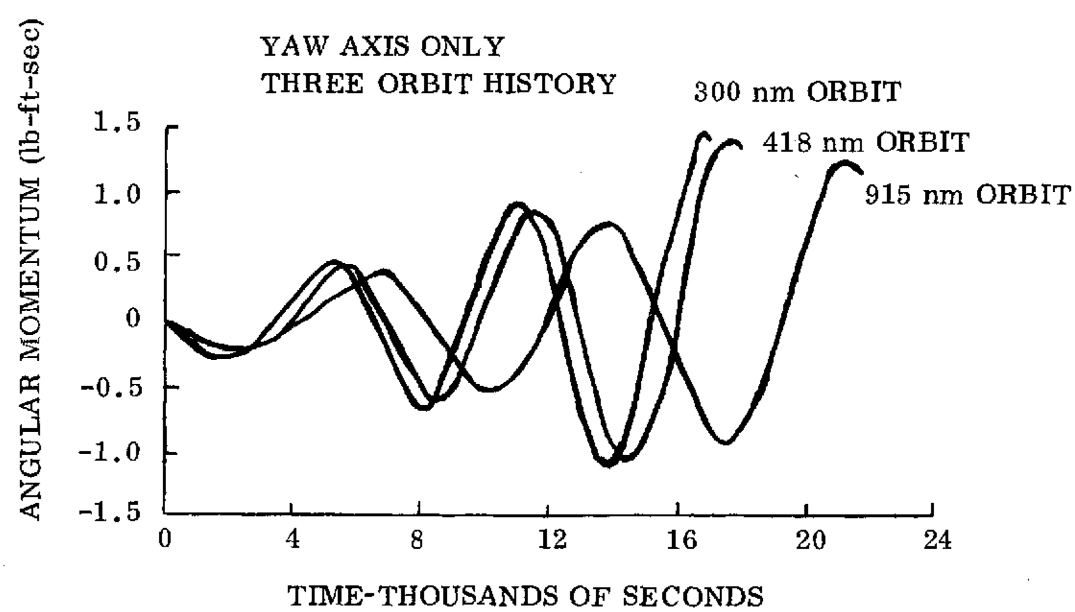

Figure 4-13. Momentum Accumulated From Magnetic Torque

The variation in momentum accumulation with altitude appears more significant than it actually is. As discussed in Section 4.3.2.2.5, the magnetic unloading subsystem also utilizes the earth's magnetic field, and its capabilities always match the momentum accumulated by magnetic torque. Hence, there is no significant trade associated with magnetic torques in an altitude trade study.

4.3.2.2.5 Magnetic Unloading Subsystem

The secular torques, and to some extent the sinusoidal torques, are unloaded by the magnetic unloading subsystem. This subsystem utilizes the earth's magnetic field which decreases with increasing altitude. Fortunately, however, magnetic, gravity gradient, and aerodynamic torques also decrease with altitude, and at an equal or faster rate. The relationship between these torques and the magnetic unloading system is therefore unchanged or improved. Solar torque represents the only disturbance torque which changes relative to the unloading subsystem. As pointed out in Section 4.3.2.2.1, however, the change is small and can easily be accommodated by the ACS designed for those altitudes.

4.3.3 ORBIT ALTITUDE EFFECTS ON THE WIDEBAND COMMUNICATION SUBSYSTEM

Wideband transmitted RF "Effective Isotropic Radiodiated Power" is directly related to orbit altitude. This relationship is shown as follows:

$$EIRP = P_t G_t = W/m^2/4KHz \times \Delta f/4000 \times 4\,h^2$$

f = spectral bandwidth

h^2 = altitude

P_t = transmitter power

G_t = antenna gain

$W/m^2/4KHz$ = power flux density in 4KHz bandwidth

Assuming power flux density to be constant and not to exceed -140 dBw/m^2/4KHz at the ground per WARC rule 740 NQ, and assuming Δ f is independent of altitude, then

$$\boxed{EIRP = P_t G_t = Kh^2 = (\text{Constant} \times \text{Altitude}^2)}$$

A performance cost trade of transmitter power and antenna gain vs. altitude is illustrated for the 240 MBs link in Table 4-4. Three discrete altitudes were used to calculate transmitter power and antenna characteristics for two cases using 500 nm as the reference altitude. In the first case a 1.7 meter diameter antenna was assumed and various transmitter powers calculated. The nearest qualified TWT amplifier was then selected. In case #2, the transmitter power was held constant and the required antenna size and beamwidth calculated. Costs were then estimated for the antenna gimbal drive.

The results are summarized in Table 4-5. The resulting cost differences assume two power amplifiers per system (only one operating at one time) and raw spacecraft power at $2K/watt. The cost data shown is only for the lowest of the cases analyzed. At the low altitude this is Case #1. At the higher altitude Case #2 is the lowest in cost because the antenna size increase is less costly than the larger power amplifier and power system costs.

Table 4- 4. Performance Cost Trade 240 MBps Link

		Case #1			Case #2			
h (nm)	Range Loss (dB)	Pt (W)	Gt (dB)	Nominal TWT Selected (watts)	Pt (W)	Gt (dB)	Antenna Diameter (M)	Antenna Bandwidth (deg)
280	165	1.3	30	1	0.4	25	0.95	9^{o}
500	170	4.	30	4	0.4	30	1.7	5.5^{o}
900	175	2.0	30	20	0.4	35	3	3^{o}

f = Const
fo = 8 GHz
Spacecraft at Nadir

EIRP = Const
Link margin 3 dB

Table 4-5. Transmitter Power and Antenna Gain Cost Trade

Altitude (nm)	Recurring Cost Difference			
	Transmitter	Antenna/Drive	Power	Total System
280	- 60K	Ref.	- 18K	- 78 K
500	Ref.	Ref.	Ref.	Ref.
900	Ref.	+20	~ Ref.	+ 20 K

4.3.4 ORBIT ALTITUDE EFFECTS ON THE THERMAL CONTROL SUBSYSTEM

The effect of altitude on the thermal design concept established is presented on Figure 4-14. where heat rejection capability is presented vs. altitude for the three subsystem modules assuming a constant Beta angle, 70^{o}F surface temperature, and degraded 5 mil Teflon over Silver Thermal coating. The heat rejection capability of the ACS module does not change with altitude; the C&DH module capability varies from -5.3% to +16.8% of nominal as the altitude changes to 300 nm and 900 nm respectively; the Power module capability varies from -2.1% to +2.8% of nominal as the altitude changes to 300 nm to 900 nm respectively. Since these differences are minor and accommodated by a slight adjustment of radiating area, the effect of altitude on the module thermal design is considered negligible. The data also shows that more than adequate heat rejection capability exists for the module areas and dissipations defined at all altitudes.

4.3.5 ORBIT ALTITUDE EFFECTS ON THE PROPULSION SUBSYSTEM

4.3.5.1 Introduction and Summary

This section presents the parametric performance analysis of the propulsion subsystem required for the EOS-A spacecraft mission. The subsystem provides to the spacecraft the propulsive functions of reaction control, orbit adjust and orbit transfer. The system design and the propellant weight requirements for performing these functions are significantly impacted by both the launch vehicle capabilities and the mission orbital altitude.

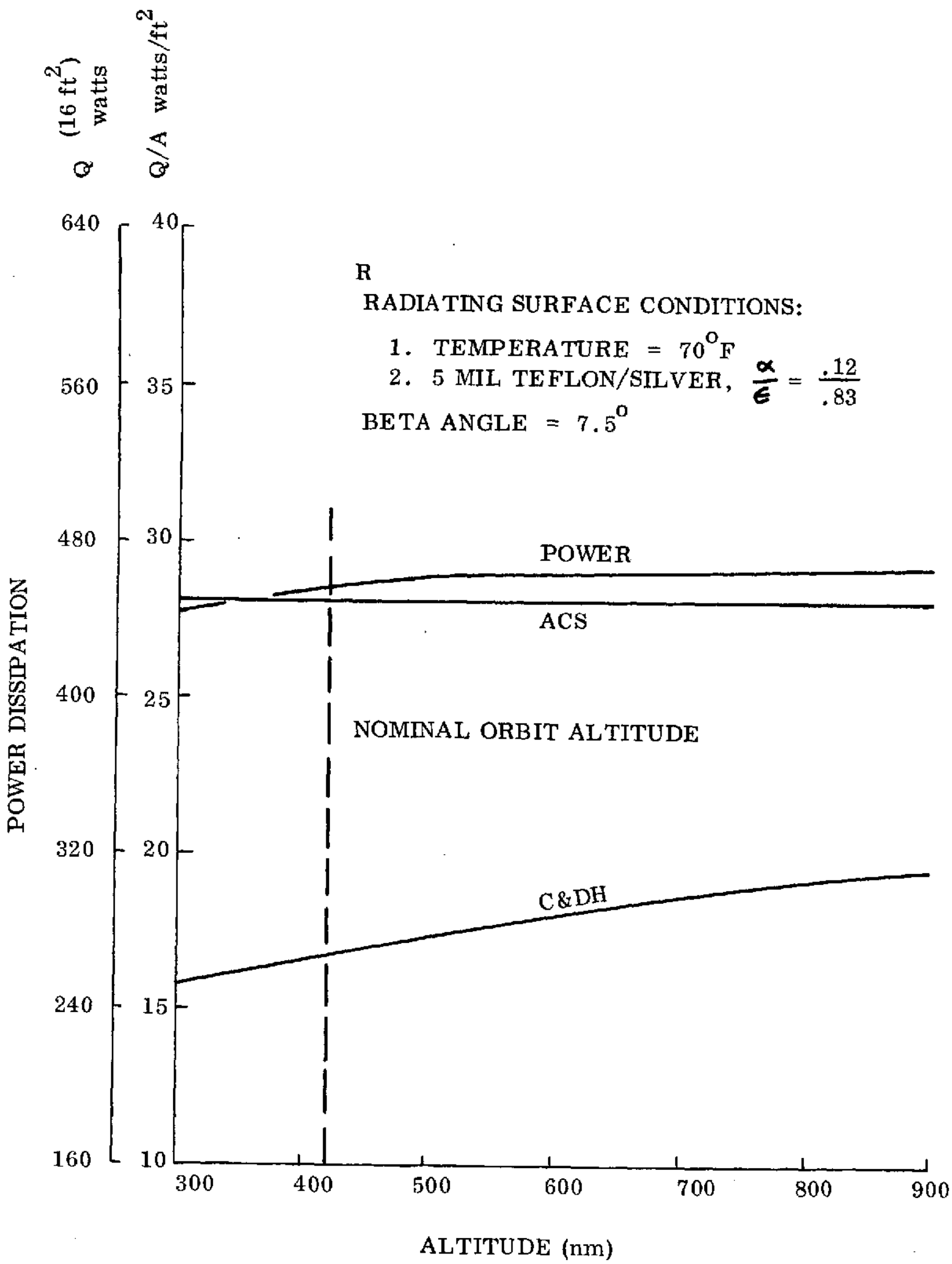

Figure 4-14. Power Dissipation as a Function of Orbital Altitude

a. Launch Vehicle Capabilities. The variables which significantly affect propulsion are:

(1) Circular vs. ellipitcal orbit injection (typical impact is presented in Figure 4-15.

(2) Spacecraft weight

(3) Spacecraft size

b. Mission Orbital Altitude. The variables affecting propulsion are:

(1) Spacecraft drag

(2) Orbit transfer for retrieval

For the purposes of this analysis, the launch vehicle candidates are limited to the Titan IIIB using no upper stage and the Delta model 2910. The basic parametric propulsion system data is presented separately for each launch vehicle. Each data set includes weights and costs as a function of mission orbital altitude. Other variables such as varying Shuttle retrieval altitudes are examined and discussed in the launch system section (Section 5.0) of this report.

4.3.5.2 Titan IIIB (NUS) Propulsion Subsystem Parametric Analysis

Because the Titan IIIB (NUS) launch vehicle does not contain a restartable upper stage, optimum payload weights are achieved by injection into elliptical low (~100 nm) perigee orbits having an apogee altitude equal to the desired mission orbital altitude. This injection scheme requires the spacecraft to contain an integral tug (orbit transfer) type propulsion system in order to circularize the orbit at the desired mission altitude. Propellant must therefore be budgeted for this function in addition to the other functions of reaction control orbit adjust, and orbit transfer (return to Shuttle for retrieval).

Figure 4-16 presents a plot of the propulsion subsystem weight required to perform these propulsive maneuvers as a function of the desired mission altitude. The assumption made in generating this plot are as follows:

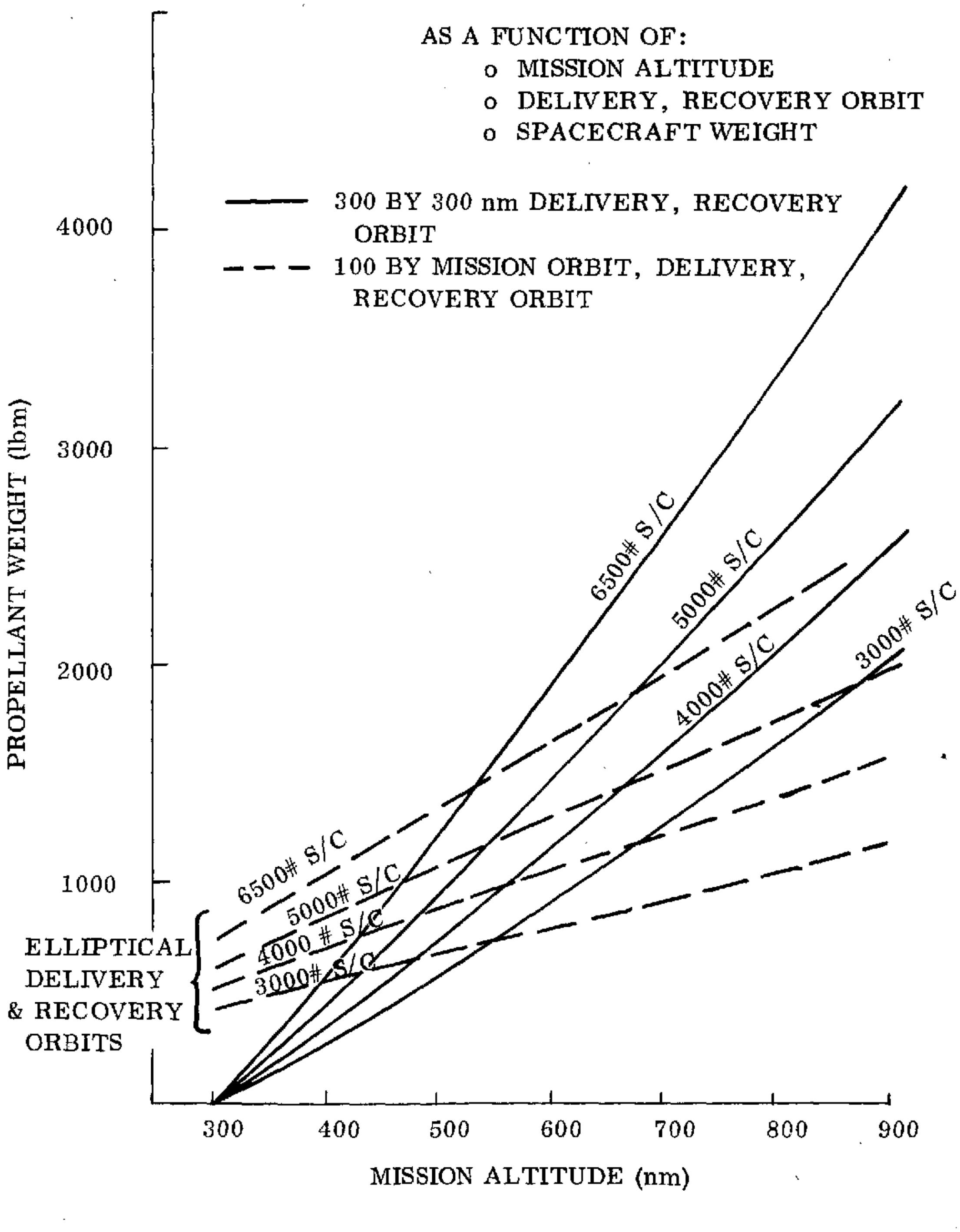

Figure 4-15. Hydrazine Propellant Weight

1. All functions are performed with an integral hydrazine propellant type propulsion subsystem.
2. Perigee of the elliptical orbit is at 100 n.
3. Retrieval by Shuttle is accomplished at a 300 nm circular orbit.
4. The spacecraft weight exclusive of the propulsion subsystem is constant at 4000 lbs.

As shown in the figure, the propulsion weight is a minimum at a mission altitude of 325 nm. This occurs because the orbit adjust fuel requirements increase as atmospheric drag becomes more significant at lower altitudes and because the orbit transfer fuel requirements increase at the higher altitudes if the spacecraft is required to be retrieved by Shuttle at the 300 nm circular altitude.

For the case of a Titan IIIB (NUS) launch, the cost of the propulsion subsystem is relatively insensitive to the mission altitude variations. Orbit transfer engines must be incorporated into the designs for all altitudes because even though a low mission altitude may negate the

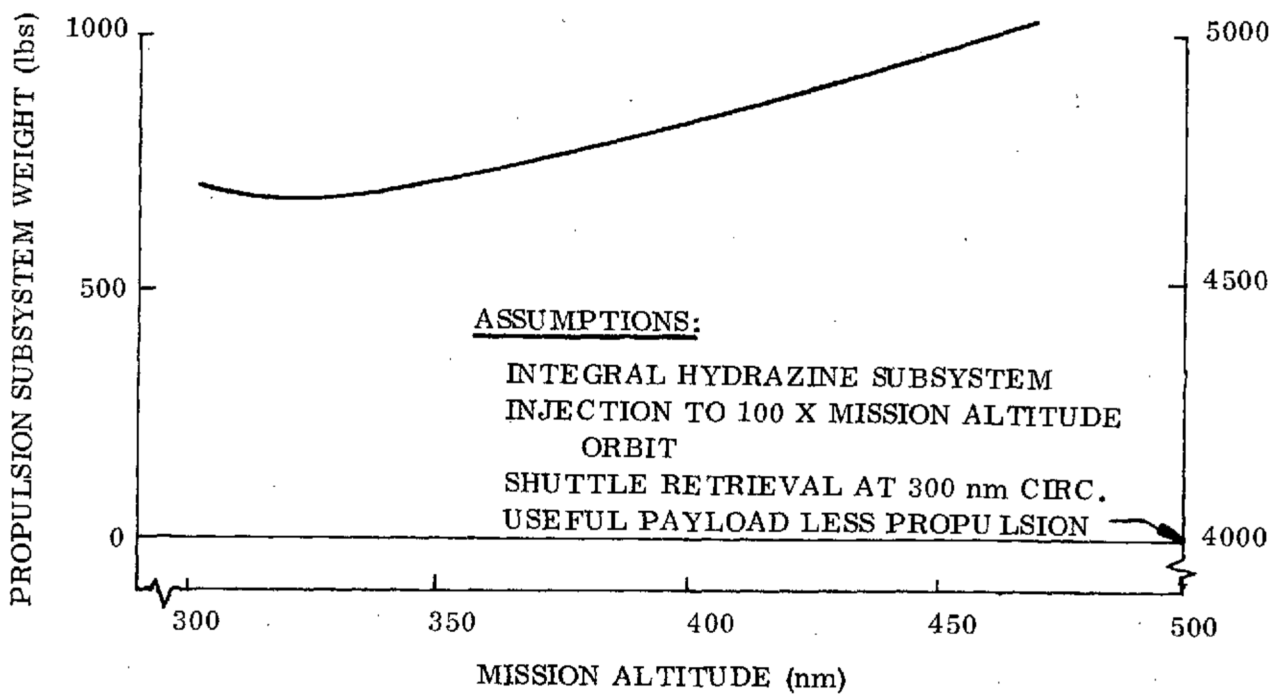

Figure 4-16. Propulsion Subsystem Weight as a Function of Mission Orbit Altitude

the requirement for retrieval orbit transfer, the launch vehicle initial orbit establishment is such that the orbit transfer is still required. Therefore the only hardware component comprising the propulsion subsystem which could be varied as a function of mission altitude is the propellant tank. For the range of propellant loads required (i.e., ~ 500 to 900 lbs) selection of a single tank size of qualified design is desired. Since all qualified designs required for this range of propellant loads have capacities in excess of 1000 lbs hydrazine when used in the blowdown mode of operation, the initial tank costs are applicable over the entire range of mission altitudes. The resultant costs for the propulsion subsystem for the Titan IIIB (NUS) launched spacecraft are as follows:

(1) Non-recurring - $2.2M
(2) Recurring - $0.7M

4.3.5.3 Delta 2910 Propulsion Subsystem Parametric Analysis

The Delta 2910 has the capability for injection into circular orbits within the mission altitude range of interest for EOS-A, therefore, the propulsion subsystem requirement for orbit transfer function is dependent solely upon the altitude capability of Shuttle for spacecraft retrieval. The cost effective altitude for Shuttle retrieval occurs within the range of 300 to 330 nm.

Figure 4-17 represents a plot of the propulsion subsystem weight required to perform spacecraft propulsive maneuvers as a function of the desired mission altitude. The assumptions made in generating this plot are identical to those presented in Section 4.3.5.2, except that spacecraft weight exclusive of the propulsion weight is assumed constant at 2200 lbs.

As shown in the figure, the propulsion system weight is near minimum at 300 nm, increases very slightly in the range of 300 to 350 nm and significantly increases at altitudes in excess of 350 nm. This results from increased orbit transfer fuel requirements at the higher altitudes since retrieval was assumed at 300 nm for all mission altitudes.

For the case of a Delta 2910 launch, the cost of the propulsion subsystem is relatively insensitive to mission altitude variations so long as the required mission altitude exceeds that of

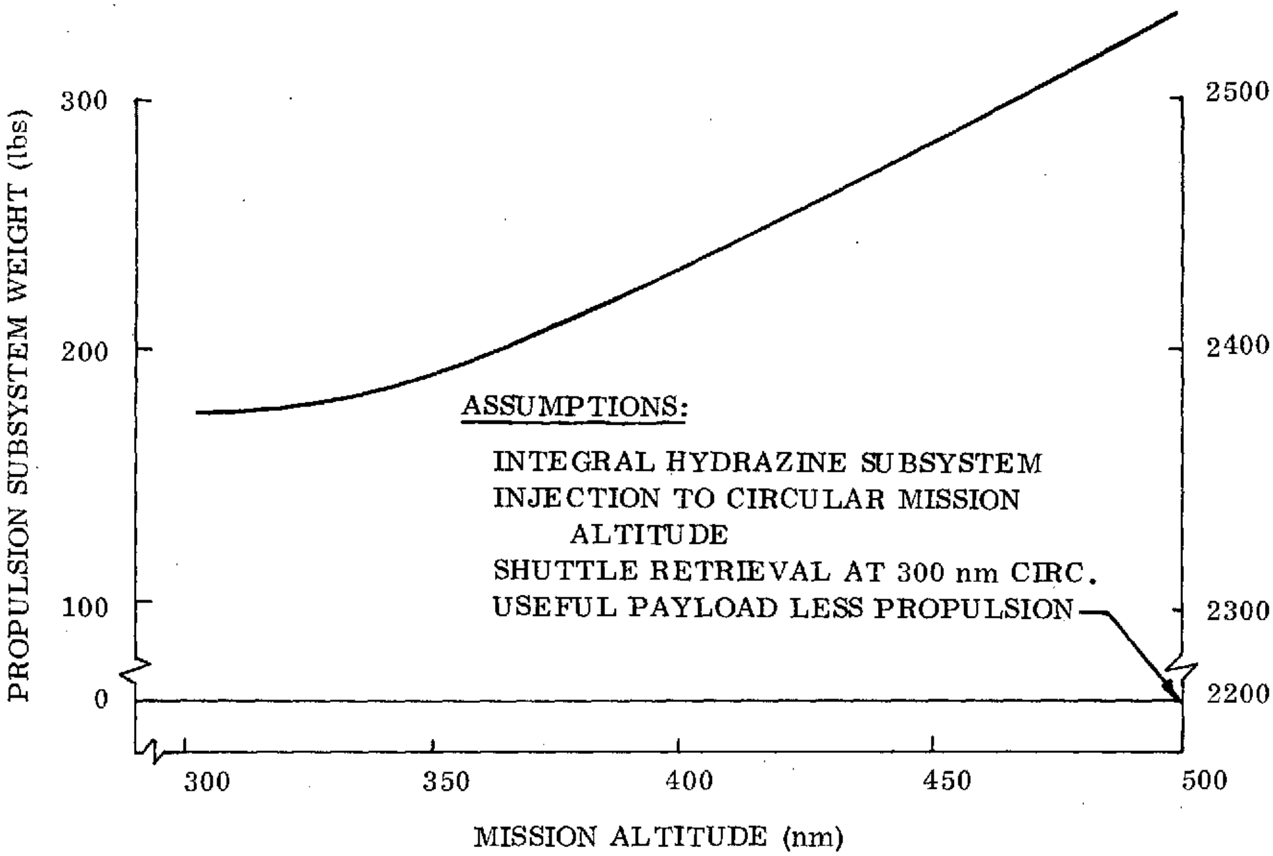

Figure 4-17. Propulsion Subsystem Weight as a Function of Mission Orbit Altitude

the Shuttle retrieval altitude. If the mission altitude is such that orbit transfer for retrieval is not required, the propulsion subsystem recurring costs are significantly reduced since the orbit transfer engines are eliminated, the number of orbit adjust engines is reduced, and propellant tankage size becomes smaller. Non-recurring costs are also reduced since the need for development and qualification of a new hydrazine engine in the thrust range of 75 to 100 lbs force is eliminated. The resultant costs for the propulsion system for the Delta launched spacecraft is contained in Table 4-6.

Table 4-6.

Propulsion Subsystem Required Functions	Non-Recurring Costs	Recurring Costs
Reaction Control & Oribt Adjust	$1.6M	$0.5M
Reaction Control, Oribt Adjust & Orbit Transfer	$1.9M	$0.6M

4.3.6.4 Orbit Altitude Effects on the Payload Instruments

Figures 4-18 and 19 represent estimates of the effect of spacecraft altitude on instrument size and weight. Each of the curves are normalized to an altitude of 775 Km and represent the relative change in the given parameter with altitude. The linear curves provide a pessimestic estimation because they assume a fixed number of detectors, in which case the optic aperture diameter varies linearly with altitude. The Te and Hughes size and weight curves are derived for an optimized design, where the number of detectors in an array (per spectral band) are varied with altitude. In all cases, the ground resolution, ground swath width and signal-to-noise ratio are held constant.

The size scale factor represents a change in optical aperture diameter which can be interpreted as instrument diameter. It can also be used to indicate a change in instrument length since the optical f/number is held constant with altitude.

Instrument costs are impacted by several parameters. The requirements that represent the major instrument cost drivers are altitude, resolution, geometric accuracy, radiometric transfer accuracy and spectral band complement. A quick survey produced no applicable general parametric instrument cost models, but a Perkin-Elmer report[1] had several curves from which a relative cost vs. optics size and figure could be derived. These curves covered such wide increments of parameter variations however, they are not very useful for costing small deviations from a nominal design.

Hughes has an object plane scanner cost model based on empirical data (MSS, VISSR, etc.) for relative cost comparisons. For a given set of mission and performance requirements (i.e., S/N, MTF) the model cost optimizes aperture size and number of detectors, which are the two major cost driving factors. It indicates (Figure 4-20) the first unit cost for an object plane scanner as a function of orbit altitude and ground resolution.

(1) "Earth Observatory Satellite - Optical Scanner Tradeoff Study, Final Report," Perkin-Elmer Report #11740, January 1974.

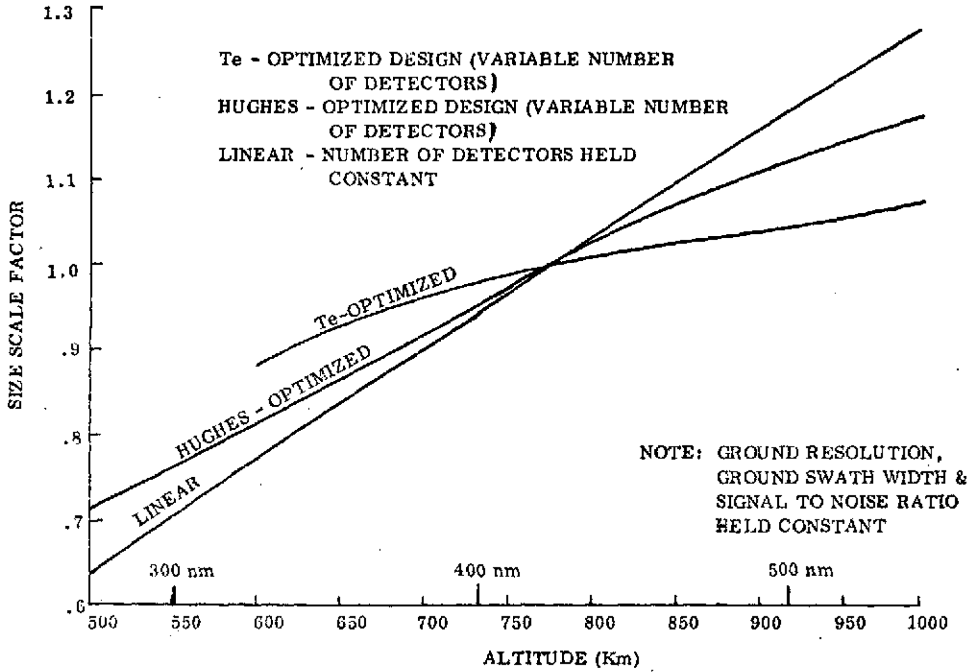

Figure 4-18. Relative Size vs. Spacecraft Altitude

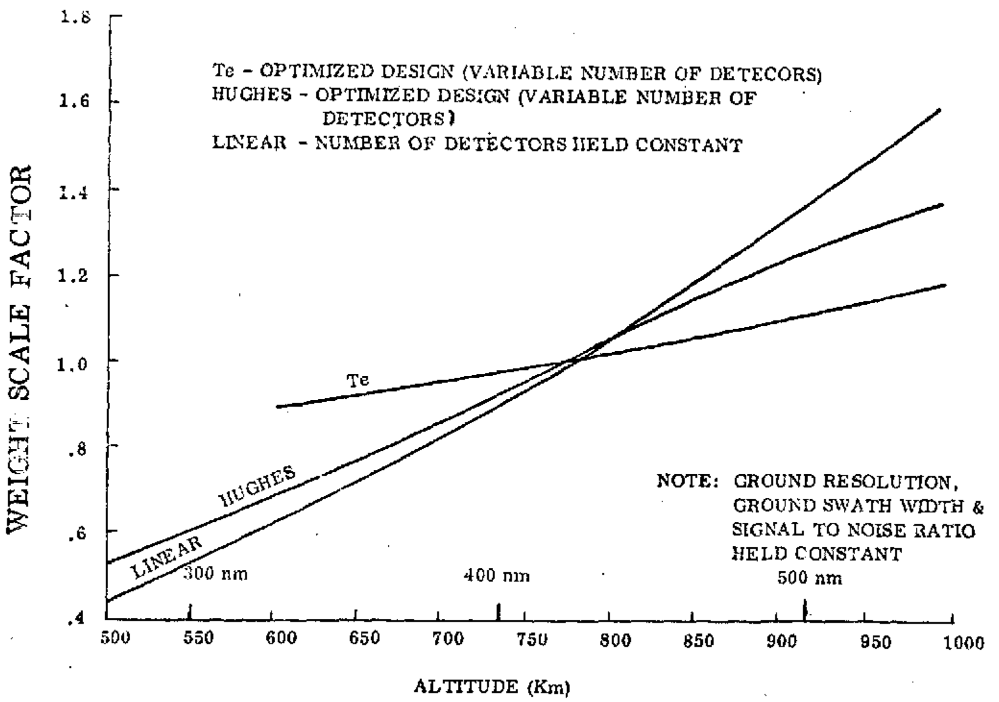

Figure 4-19. Relative Weight vs. Spacecraft Altitude

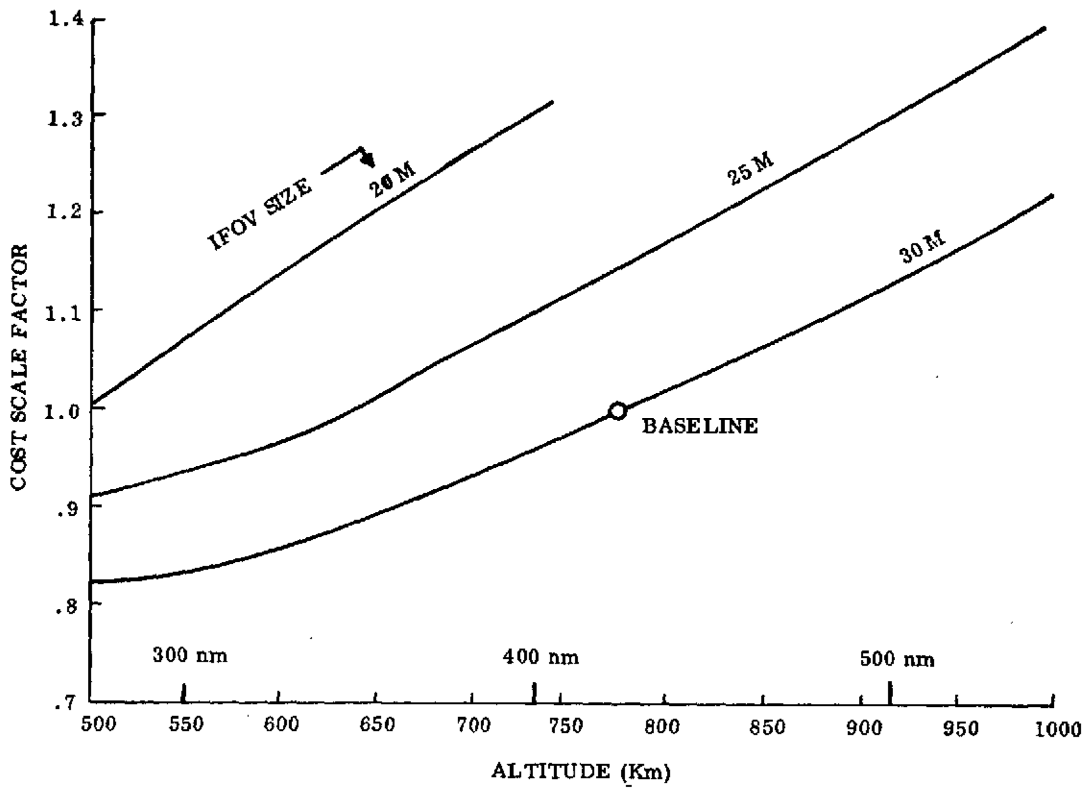

Figure 4-20. Relative Cost vs. Spacecraft Altitude

The Hughes curves are smooth parametric ones. Actual instrument cost would be tempered by the existance of developed designs and equipment. For example, it is likely that designs would be based on a few discrete aperture sizes. Performance then, would be allowed to vary over small altitude ranges. This would cause stepped cost curves, which would be relatively flat over a small (perhaps $\pm$50 nm) altitude change.

Although no other parametric cost curves were available from other instrument constractors, verbal discussion also indicates the practicality of accepting performance deviations over a range of altitudes while retaining the same base optical/mechanical scan mechanism. Thus the parametric curves of Figure 4-20 represents a very pessimestic view of cost vs. altitude for modest altitude changes, but a reasonable estimate over large altitude changes.

SECTION 5.0
LAUNCH SYSTEM PARAMETRIC ANALYSIS

5.1 INTRODUCTION AND SUMMARY

This section contains the parametric performance analysis of the EOS-A launch system which includes the launch vehicle, integral propulsion system and Shuttle retrieval system. The integral propulsion system is included in the parametric analysis since it plays an integral role in achieving the mission orbit when Titan launch vehicles are considered and also provides the capability of retrieving the spacecraft by Shuttle at an altitude other than the mission altitude.

The low cost launch system recommended for EOS-A (TM and HRPI) includes:

(1) A Delta 2910 launch vehicle
(2) An integral propulsion system (hydrazine) to return the spacecraft to Shuttle at 330 nm for retrieval.

This system gives an overall transportation system cost (Delta - Non-Recurring and Recurring, Propulsion System - Non-Recurring and Recurring, and Shuttle retrieval) of approximately $ 12M compared to a cost of greater than $19M for a Titan IIIB integral tug system and a cost in excess of $ 40M for Titan IIID integral tug.

5.2 INTEGRAL PROPULSION SYSTEM

A complete tradeoff analysis of alternate propulsion systems is presented in Report #3 "Design/Cost Tradeoffs" which reaches the conclusion that a hydrazine propulsion system is preferred for the EOS integral propulsion system. Table 5-1 summarizes the tradeoff analysis showing the all hydrazine system is lowest cost for EOS-A (non-recurring cost and one flight unit) and also lowest cost for total program containing four flight units which are refurbished for an additional ten flights.

The total trade summary shown at the bottom of Table 5-1, which rates the three alternate systems against ten criteria, clearly indicates the superiority of the all hydrazine system.

Table 5-1. Propulsion System Trade Summary

Costing Assumptions

NR = includes qual unit
REC = four flight units
REF = refurbish flight units for 10 additional flights

Cost Summary

Titan Spacecraft

Design	Non-Rec	Costs in M$ Rec.	Refurb.	EOS-A One Flight	Total Program
NASA Baseline					
Boeing	5.000	0.650	0.400	5.650	11.6
GE	2.865	1.087	0.400	3.952	11.2
Lowest	2.865	0.650	0.400	3.515	9.5
Baseline Variations (N_2H_4 and Solids)	2.202	0.897	0.350	3.102	9.3
Alternate (all N_2H_4)	2.160	0.680	0.120	2.840	6.1

Delta Spacecraft
(with RCS, OA & OT)

Design	Non-Rec	Rec.	Refurb.	EOS-A One Flight	Total Program
NASA Baseline Variation (GN_2 & N_2H_4)	2.405	0.690	0.11	3.095	6.3
Alternate (all N_2H_4)	1.935	0.600	0.10	2.535	5.3

Trade Summary

Titan Spacecraft

Evaluation Criteria	Design Configuration NASA Baseline	Baseline Variation	Alternate
System Cost	3	2	1
System Weight	3	1	2
Mission Flexibility	2	2	1
Growth Potential	2	2	1
Development Risk	1	1	1
Reliability and Simplicity	2	2	1
Shuttle Compatiblity	1	1	1
Design Modularity	1	1	1
System Safety	1	1	1
Vehicle Design Impacts	2	2	1
Overall Rank	3	2	1

The mission flexibility inherent in the hydrazine system has a side benefit in simplifying the parametric performance analysis of the total launch system.

5.3 DELTA 2910 PARAMETRIC ANALYSIS

5.3.1 LAUNCH VEHICLE PERFORMANCE

The Delta 2910 launch vehicle injection capability into sun-synchronous circular orbits having altitudes ranging from 300 to 500 nm is presented in Figure 5-1. As shown in the figure, the capability to 300 nm is 2820 lbs and decreases along a near-linear curve to 2440 lbs at 500 nm. Also presented in this figure are curves of useful spacecraft weight for the following two retrieval assumptions:

- o Retrieval accomplished at mission altitude
- o Retrieval accomplished at an orbit circular altitutde of 300 nm

The useful spacecraft weight is defined as the difference between the launch vehicle injection capability and the weight of a propulsion subsystem having the capability of performing all spacecraft propulsive functions required during a three year EOS mission. The propulsion subsystem weight was calculated based upon the following assumptions:

1. The subsystem type is of integral hydrazine design having the capability for reaction control, orbit adjust and transfer.
2. All in-plane and cross-track launch vehicle injection errors are removed.
3. Orbit maintenance is based upon a spacecraft having a drag area of 140 ft^2.

The difference between the two useful weight curves represents the weight of propellant required to affect the Hohlmann transfers to a 300 nm retrieval orbit plus the weight of tankage required to contain this propellant and the weight of two (redundant) orbit transfer engines. The difference between the capability curve and mission altitude retrieval curve represents the propulsion subsystem weight required to perform the reaction control and orbit adjust functions. The increased weight requirement at lower mission altitudes reflects the additional propellant weights expended to counter the effects of atmospheric drag upon the spacecraft.

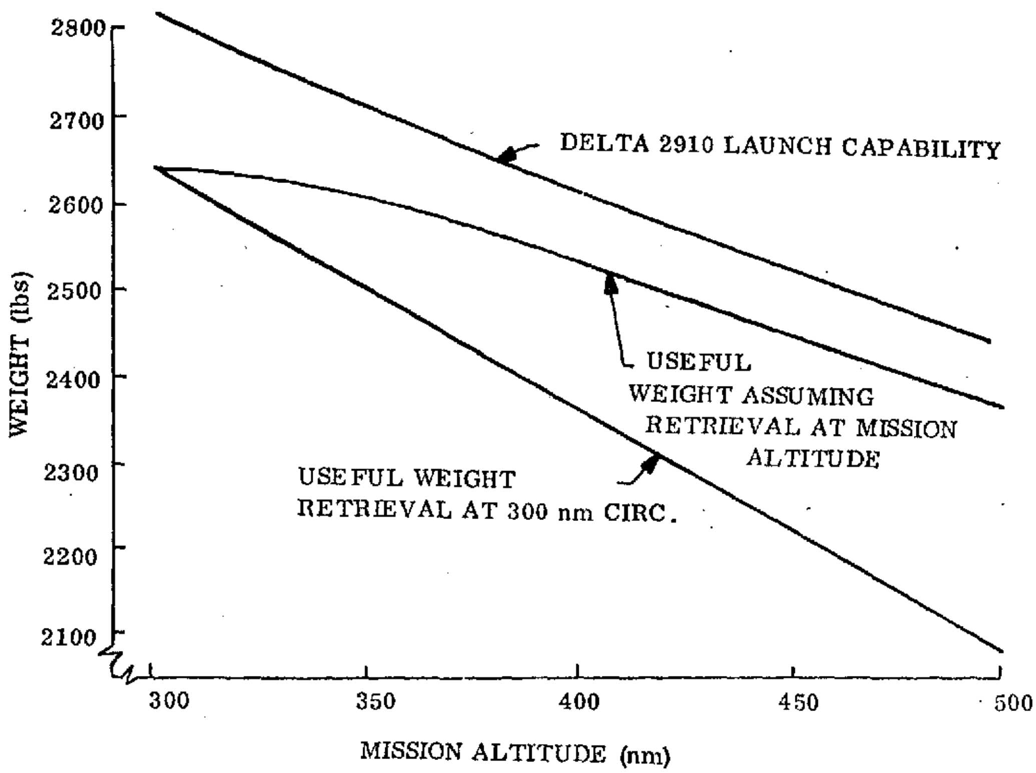

Figure 5-1. Delta 2910 Launch System Performance

Examination of the useful payload curves of Figure 5-1 for the case of a 2200 lb EOS-A spacecraft (representing the current predicted spacecraft weight) reveals the following:

1. If retrieval is accomplished at 300 nm, the range of available mission altitudes for the 2910 launched system is limited to a maximum of 460 nm.
2. If retrieval is accomplished at mission altitude, or if retrieval is not accomplished, the 2910 Delta has sufficient capability over the entire range of 300 to 500 nm altitudes. (Shuttle cannot retrieve above 430 nm.)

5.3.2 RETRIEVAL ALTITUDE IMPACTS

The impact of various end-of-mission retrieval altitudes was studied for a 2910 Delta launched spacecraft. The data presented in Figure 5-2 are for an assumed mission altitude of 418 nm. As shown in the figure, the Delta 2910 capability to the orbit is 2580 lbs. The useful payload curve is again the difference between launch capability and the weight of the

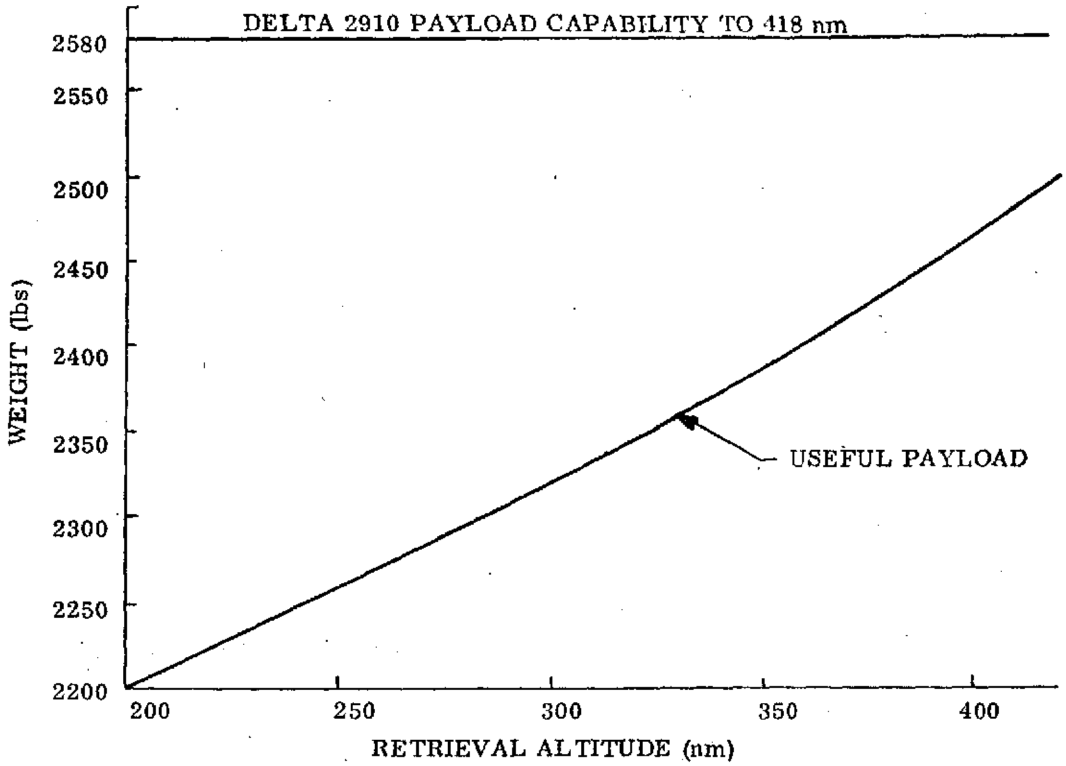

Figure 5-2 . Delta 2910 Useful Payload as a Function of Shuttle Retrieval Altitude

propulsion subsystem which is capable of performing the mission reaction control, orbit adjust and orbit transfer functions. The figure shows that for the 2200 lb EOS-A spacecraft operating at 418 nm, the minimum retrieval altitude which can be accomplished is at 200 nm circular.

5.3.3 LAUNCH SYSTEM COSTS

Costs for a propulsion subsystem compatible with a 2910 Delta launched EOS spacecraft are presented in Table 5-2. Costs are presented for two designs, one assuming retrieval at mission altitude and the other assuming a 300 nm retrieval. Various combinations of gaseous nitrogen and liquid hydrazine types of systems were costed, however, the integral hydrazine type exhibited the lowest cost for both retrieval cases.

The Delta 2910 launch vehicle recurring costs are quoted at \$6.0M which, when added to the propulsion subsystem costs, results in a launch system cost of \$6.45M for a spacecraft retrieved at mission altitude and \$ 6.6 M for one retrieved at 300 nm.

5.4 DELTA 3910 PARAMETRIC ANALYSIS

The Delta 3910 launch vehicle injection capability into sun-synchronous circular orbits exceeds that of the 2910 Delta by approximately 1000 lbs. As shown in Figure 5-3, the capability to 300 nm is 3890 lbs. and decreases along a near-linear curve to 3420 lbs. at 500 nm. The figure also presents plots of useful payloads for the cases of mission altitude retrieval and a 300 nm retrieval. All assumptions made in establishing these curves are the same as those defined in Paragraph 5.3. These payloads are in the range of 3000 to 3700 lbs. depending upon the retrieval and mission altitudes.

The useful payload as a function of retrieval altitude was also examined for the 3910 Delta launched spacecraft. A plot of these data for a mission altitude of 418 nm is contained in

Table 5-2. Propulsion System Trade Summary for a Delta Launched EOS Spacecraft

Costing Assumptions

NR = includes qual unit
REC = four flight units
REF = refurbish flight units for 10 additional flights

Design	Cost in M\$				
	Non-Recurring	Recurring	Refurbish	EOS-A One Flight	Total Program
Retrieval at Mission Altitude					
All Gaseous Nitrogen	1.510	.460	.065	1.970	4.000
Gas Nitrogen + Hydrazine	2.005	.540	.080	2.545	4.965
Integral Hydrazine	1.530	.450	.065	1.980	3.980
Retrieval at 300 nm Altitude					
Gas Nitrogen + Hydrazine	2.405	.690	.110	3.095	6.265
Integral Hydrazine	1.935	.600	.100	2.535	5.335

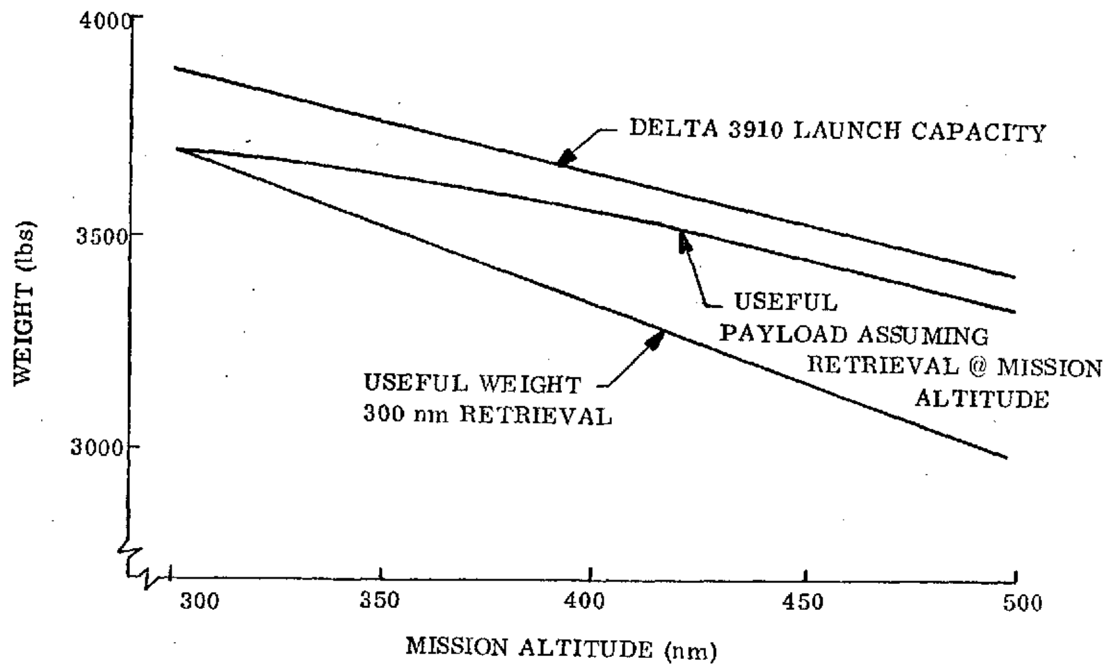

Figure 5-3. Delta 3910 Launch System Performance

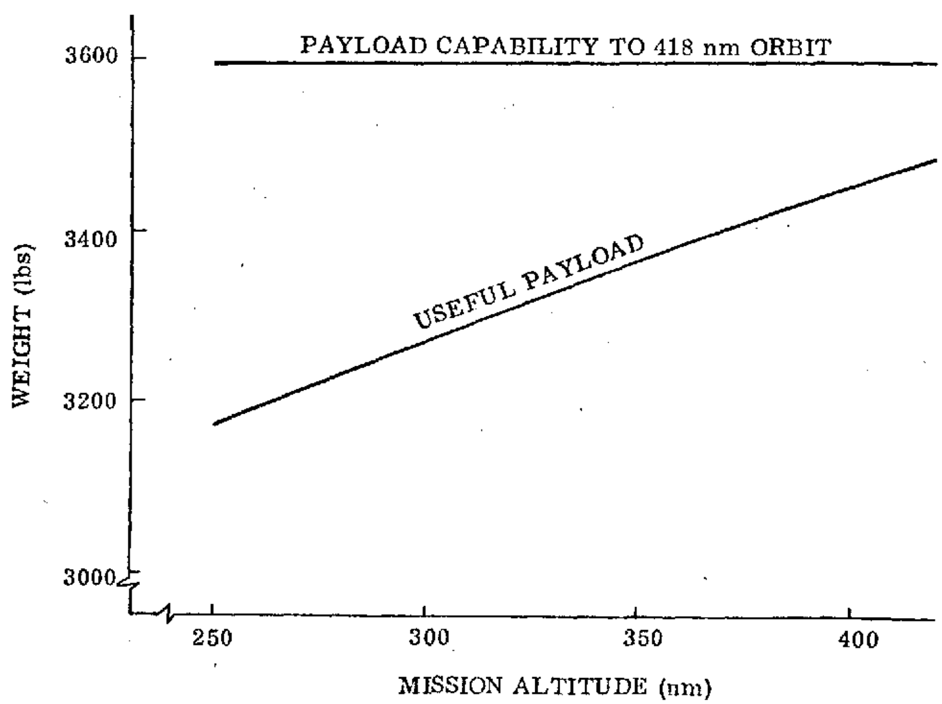

Figure 5-4. Delta 3910 Useful Payload as a Function of Shuttle Retrieval Altitude (Mission Altitude = 418 nm)

Figure 5-4. The useful payload curve parallels that for the 2910 Delta but allows a spacecraft weight of 900 to 1000 lbs. higher than that of the 2910 Delta.

The Delta 3910 launch vehicle recurring costs are quoted at $ 8.0M. The propulsion subsystem costs associated with this launch vehicle are identical to those for the 2910 Delta resulting in a launch system cost of $8.6M for the 3910 Delta.

5.5 TITAN IIIB INTEGRAL PROPULSION PARAMETRIC ANALYSIS

The Titan IIIB launch vehicle without an upper stage has extremely limited injection capability to circular orbits having altitudes above 100 nm but good capability to elliptical orbits having a perigee in the range of 85 to 100 nm. The injection capability into these elliptical orbits is shown in Figure 5-5 and ranges from 5400 lbs. into a 100 x 300 nm orbit to 5000 lbs. into a 100 x 500 nm orbit.

Using the elliptical orbit injection scheme requires an integral propulsion system for circularizing the orbit at the desired mission altitude. This propulsion system also supplies the reaction control and orbit functions for a three year mission. Other assumptions made in calculating the propulsion system weight are as follows:

1. The system uses hydrazine propellant for accomplishing all propulsive functions.
2. All in-plane and cross-track launch vehicle injection errors are removed.
3. Orbit maintenance is based upon a spacecraft having a drag area of 280 ft^2.

The propulsion subsystem weight capable of performing these functions is shown by the difference between the launch vehicle capability curve and the useful payload curve which assumes retrieval at mission altitude. If retrieval is to be accomplished at 300 nm, the propulsion subsystem weight is increased by the difference between the two useful payload curves. The curve shows that over the 300 to 500 nm mission altitude range, the Titan IIIB launch system has a useful payload capability ranging from 4680 to 4290 lbs. for the mission altitude retrieval case and from 4680 to 3900 lbs. for the 300 nm retrieval case.

The impact of various end-of-mission retrieval altitudes was studied for the Titan IIIB launched spacecraft. The data presented in Figure 5-6 are applicable for an assumed mission altitude of 418 nm. The difference between the payload capability curve and the useful payload curve represents the weight of the spacecraft integral propulsion subsystem.

Costs for various propulsion subsystem designs capable of performing the propulsive functions associated with a Titan IIIB launched EOS spacecraft are shown in Table 5-3. The system having the lowest program costs, though not the lowest recurring costs, is the integral all-hydrazine propellant design. Recurring costs for this design are shown as \$680K. Adding this to the quoted Titan IIIB launch vehicle costs of \$8.5M results in a launch system cost of \$9.18M. This cost is independent of the retrieval altitude since the Titan IIIB launched spacecraft propulsion system must perform the orbit transfer function for initial orbit establishment.

5.6 TITAN IIID - INTEGRAL TUG

The Titan IIID - Integral Tug performance into the range of orbits being considered for EOS-A far exceeds the performance of the Delta and Titan IIIB. Since the Titan IIIB performance is adequate for all projected missions including resupply and the cost of the Titan IIID is more than double the cost of the Titan IIIB, the Titan IIID system has been eliminated from further consideration. No further studies will be made of the Titan IIID.

5.7 SHUTTLE PARAMETRIC PERFORMANCE ANALYSIS

The space Shuttle payload capability into sun-synchronous orbits is shown in Figure 5-7. Four curves are shown in the figure with the upper curve defining the delivery capability (assuming no rendezvous). The second curve defines the payload capability when a rendezvous in orbit is required. Both of these curves were extracted from the Space Shuttle System Payload Accommodations document, JSC-07700 Volume XIV, Revision B, dated December 21, 1973 and present round trip capabilities for the Shuttle. The two lower curves were generated to establish allowable spacecraft weights by subtracting the flight support system weight (assumed as 1500 lbs. for eary EOS recoveries) and the resupply system weight (assumed as 2200 lbs.).

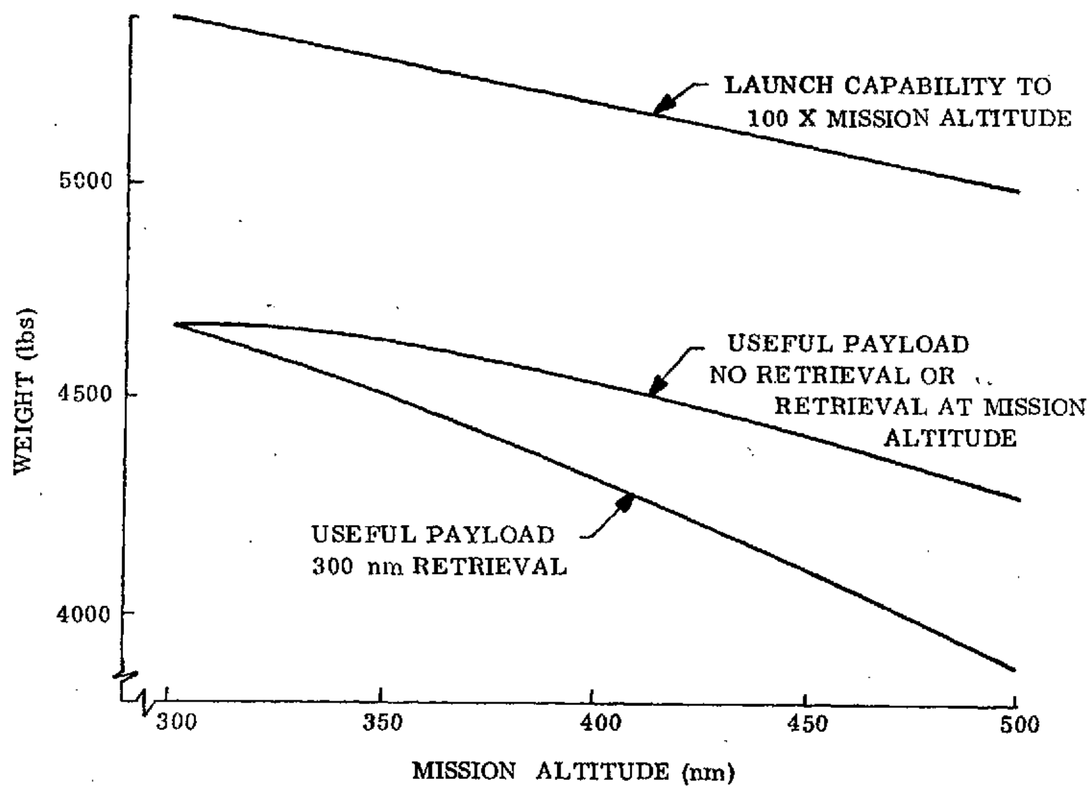

Figure 5-5. Titan IIIB (NUS) Launch System Performance

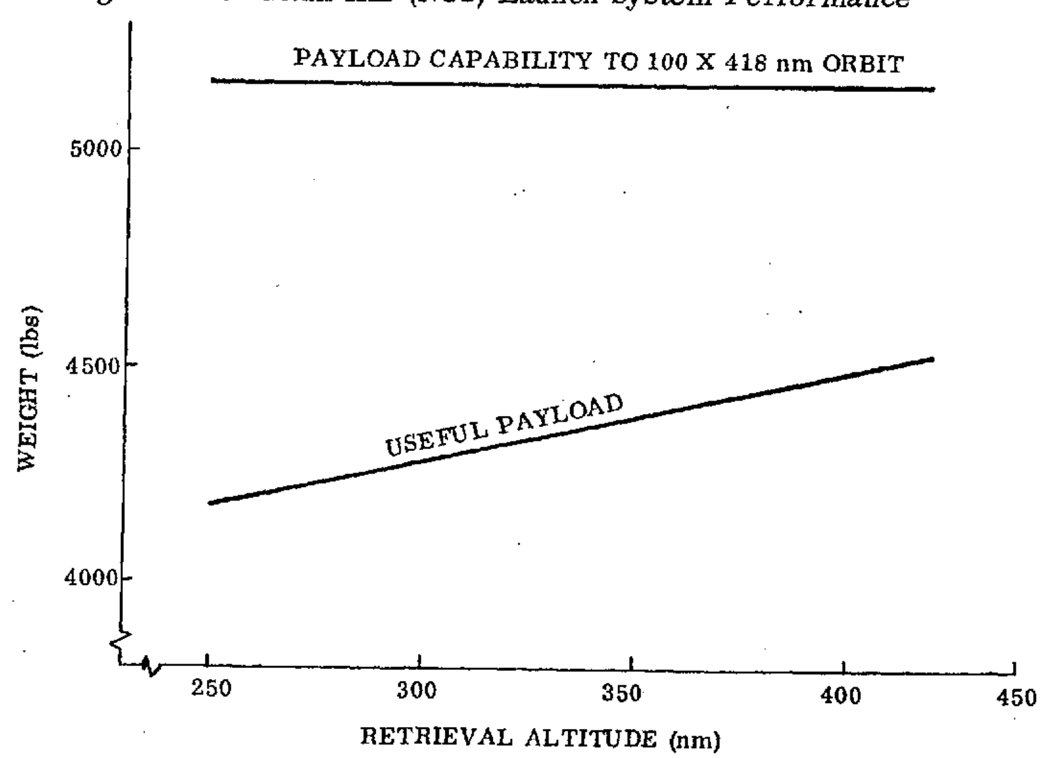

Figure 5-6. Titan IIIB (NUS) Useful Payload as a Function of Shuttle Retrieval Altitude (Mission Altitude = 418 nm)

Table 5-3. Propulsion System Trade Summary for Titan IIIB Launched EOS Spacecraft

Costing Assumptions

NR = includes qual unit
REC = four flight units
REF = refurbish flight units for 10 additional flights

Cost Summary

Design	Non-Recurring	Cost in M$ Recurring	Cost in M$ Refurbish	EOS-A One Flight	Total Program
NASA Baseline					
Boeing	5.000	0.650	0.400	5.650	11.6
GE	2.865	1.087	0.400	3.952	11.2
Lowest	2.865	0.650	0.400	3.515	9.5
Baseline (Variation) (N_2H_4 and Solids)	2.202	0.897	0.350	3.102	9.3
Alternate (all N_2H_4)	2.160	0.680	0.120	2.840	6.1

Trade Summary

Evaluation Criteria	Design Configuration: NASA Baseline	Design Configuration: Baseline Variation	Alternate
System Cost	3	2	1
System Weight	3	1	2
Mission Flexibility	2	2	1
Growth Potential	2	2	1
Development Risk	1	1	1
Reliability & Simplicity	2	2	1
Shuttle Compatibility	1	1	1
Design Modularity	1	1	1
System Safety	1	1	1
Vehicle Design Impacts	2	2	1
Overall Rank	3	2	1

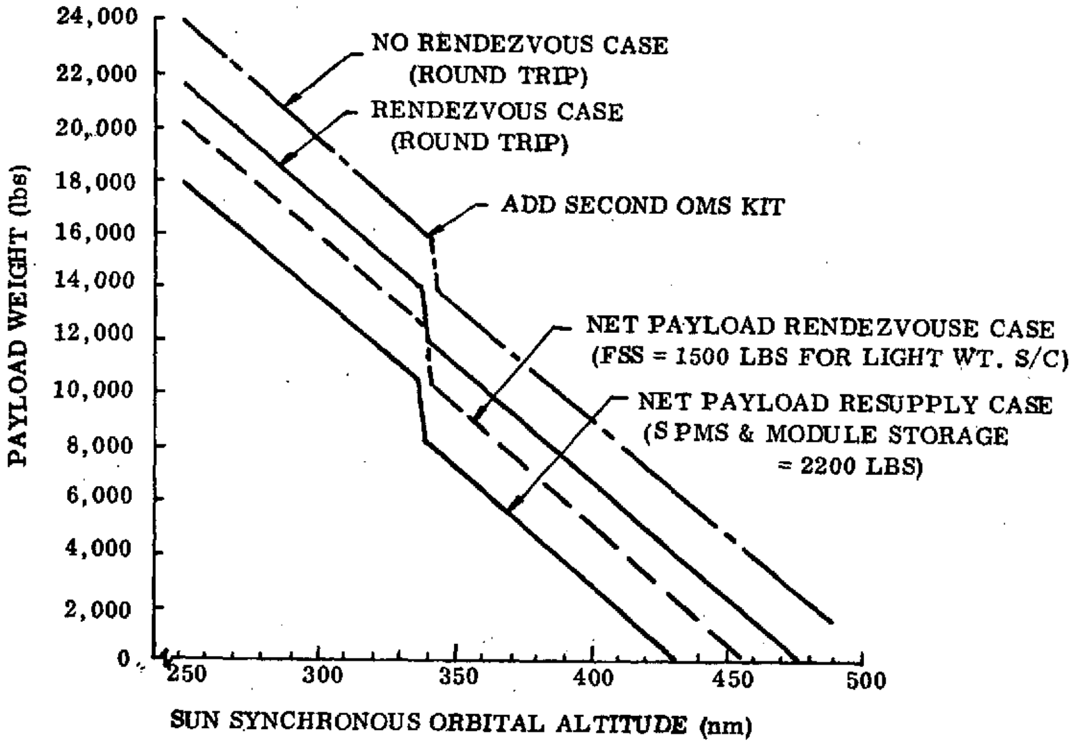

Figure 5-7. Shuttle Payload Weight vs. Sun-Synchronous Orbital Altitude

The curves indicate that the maximum altitude for Shuttle retrieval of a Titan spacecraft weighing 380 lbs. is approximately 415 nm while the maximum retrieval altitude of a Delta spacecraft weighing 2200 lbs. is approximately 430 nm. An additional item of interest in Figure 5-7 is the step down in the Shuttle performance that occurs between 330 and 340 nm due to the addition of the second OMS tank. If Shuttle retrieval altitude is limited to 330 nm or lower, the second OMS tank is not required and Shuttle performance is significantly better allowing a more cost effective sharing of the Shuttle capability with other payloads.

The Shuttle cost ground rules given in Table 5-4 show a maximum Shuttle two way cost of \$9.8M. Also the Shuttle costs up and down are defined as a ratio of the Shuttle weight capability used. For the purposes of this early study it will be assumed that only 78% of the allowable Shuttle payload can be loaded into the cargo bay due to volume restrictions, cargo availability, etc. This assumption will increase the percent of Shuttle charges that

EOS must pay while not requiring detail analysis of alternate cargo manifests which would unduly complicate early parametric studies. The 78% Shuttle loading factor was selected by analysing the present Shuttle traffic model while factoring in experience GE has gained on recent Shuttle payload studies.

5.8 OOS/TUG PERFORMANCE

The SEOS mission which is a geosynchronous earth observatory requires a OOS or Tug to provide the Δ V to transfer the spacecraft from the Shuttle orbit to the mission orbit. The OOS is an interim tug with delivery only capability which will be replaced by the full capability tug in 1983. The performance and cost data for both tugs is summarized in Table 5-5.

Table 5-4. Shuttle Costs

Payload Up and Down Cost = 9.8M Max (4.9M Up and 4.9M Down)

Payload Up Cost = 4.9M (Load Factor)

$$\text{Where Load Factor} = \frac{\text{Payload Up Weight}}{\text{Shuttle Payload Capability}}$$

Payload Down Cost = 4.9M (Load Factor)

$$\text{Where Load Factor} = \frac{\text{Payload Down Weight}}{\text{Shuttle Payload Capability}}$$

Cargo Manifest - Share Payloads

- o Materials Processing Module
- o Life Sciences Module
- o Short Pallet
- o Hitch Hiker Pallet

Table 5-5. Tug Performance

Type Tug	Availagle Date	Delivery	Retrieve	Round Trip	Cost
OOS	1980	3600	--	--	TBD
TUG	1983	7200	5400	3400	TBD

5.9 LAUNCH SYSTEM COST TRADES

Launch system cost trades have been investigated as a function of launch vehicle, mission altitude and Shuttle retrieval altitude for EOS-A. The launch system costs used in these trades are shown in Table 5-6.

Table 5-6. Launch System Costs

Launch Vehicle Costs	Recurring	Non-Recurring
Delta 2910	6M	2M
Delta 3910	8M	2M
Titan IIIB NUS	12.2M	3M
Propulsion System Costs		
RCS & OA	.5M	1.6M
Delta, RCS, OA & OT	.6M	1.9M
Titan, RCS, OA & OT	.7M	2.2M

Using these launch system costs and the Shuttle retrieval costs discussed in Section 5.7 and shown in Table 5-4, an analysis of EOS-A transportation cost was established to determine the cost effective launch system for EOS-A. A summary of this study is shown in Tables 5-7 and 5-8. Three Delta 2910 launch/retrieval options are shown. The first assumes a mission altitude of 360 nm (generally considered the lowest feasible mission altitude) with retrieval of Shuttle at that altitude (no orbit transfer system required on the spacecraft). The second Delta 2910 option shows a mission altitude of 420 (preferred from mission viewpoint) with an orbit transfer system added to the spacecraft to bring the spacecraft back to 330 nm for Shuttle retrieval. The second case shows that there is no transportation cost penalty if the preferred mission orbit is selected and an orbit transfer system is added to bring the spacecraft to a more economical Shuttle retrieval orbit (the cost of the orbit transfer system is more than offset by the Shuttle retrieval cost savings). The third case showing the cost if the Shuttle retrieves the spacecraft at a mission altitude of 420 nm (with a cost penalty of $2.6M over Option 2) also illustrates the cost advantages of adding orbit transfer capability to the spacecraft propulsion system. The remaining two options in Table 5-7 show the total transportation cost penalty associated with Delta 3910 and Titan IIIB launches. The penalty for a Delta 3910 launch is not prohibitive since it

Table 5-7. EOS-A Transportation Costs Trade Summary

Option No.	Description of Option	Delivery Cost (M$) NR	Delivery Cost (M$) REC	Retrieve Cost (M$)	Total Cost (M$)	Allowable S/C Wt (lbs.)
1	o Launch with Delta 2910 to 360 nm o Retrieve with Shuttle @ 360 nm	3.6	6.5	2.3	12.4	2590
2	o Launch with Delta 2910 to 420 nm o Retrieve with Shuttle @ 330 nm	3.9	6.6	1.6	12.1	2360
3	o Launch with Delta 2910 to 420 nm o Retrieve with Shuttle @ 420 nm	3.6	6.5	4.6	14.7	2490
4	o Launch with Delta 3910 to 420 nm o Retrieve with Shuttle @ 250 nm	3.9	8.6	1.1	13.6	3175
5	o Launch with Titan IIIB Integ. Tug to 420 nm o Retrieve with Shuttle @ 250 nm	5.2	12.9	1.4	19.5	4180

allows an additional spacecraft weight of approximately 800 lbs. Sample calculations for the Delta 2910 Options 1 and 2 are shown in Table 5-9 to illustrate how the transportation costs in Table 5-7 were derived. A plot of EOS transportation costs is presented in Figure 5-8 showing the impact of selecting a wide range of mission altitudes and Shuttle retrieval altitudes. The upper curve showing total transportation costs as a function of Shuttle retrieval altitude is independent of selected mission altitude, therefore, a low transportation cost of approximately $12M can be achieved using an integral propulsion system independent of mission altitude as long as the orbit transfer fuel weight required plus the spacecraft weight falls within the performance capability of the launch vehicle. As shown on the right hand side of Table 5-7, all the options considered are compatible with a light weight EOS-A weighing approximately 2200 lbs. The data presented in Table 5-7 can be expanded to include the spacecraft cost per pound and the allowable payload instrument weight as presented in Table 5-8. Option #2 presents the lowest transportation costs of $12.1M while restricting the allowable spacecraft weight and payload weight to 2360 lbs. and 650 lbs. respectively. If the instrument payload weight exceeds the allowable

Table 5-9. Sample Transportation Cost Trades for Delta 2910 EOS-A

OPTION #1

- Launch to Shuttle compatible orbit = 360 nm with Delta 2910
- Retrieve with Shuttle@360 nm

EOS-A Retrieval wt = 2200#

Shuttle Supt wt. = 1500#

Delivery Cost	NR	R
Delta Launch Cost	2.0	6.0
Prop. System Cost	1.6	0.5
Retrieval Cost		
Shuttle Capability = 10,200#		
Cost = $\$4.9M \left(\frac{3,700}{10,200}\right)\left(\frac{1}{.78}\right)$		2.3
Total Trans. Cost =	3.6 +	8.8 = $12.4M

OPTION #2

- Launch to Shuttle compatible orbit = 420 nm with Delta 2910
- Retrieve with Shuttle@330nm

EOS-A Retrieval wt = 2200#

Shuttle Supt, wt = 1500#

Delivery Cost	NR	R
Delta Launch Cost	2.0	6.0
Prop. System Cost	1.9	0.6
Retrieval Cost		
Shuttle Capability = 14,600#		
Cost = $\$4.9M \left(\frac{3,700}{14,600}\right)\left(\frac{1}{.78}\right)$		1.6
Total Trans. Cost =	3.9 +	8.2 = $12.1M

payload weight of the Delta 2910 at 420 nm, three solutions are available which are to:

- o Eliminate Shuttle retrieval (200 lbs. increased payload capability, transportation cost savings of $2.0M, and loss of all hardware for later use).
- o Reduce the mission altitude to 360 nm and retrieve directly by Shuttle (230 lbs. increased payload capability at an increased transportation cost of $300K).
- o Substitute Delta 3910 for Delta 2910 and retrieve at 250 nm (510 lbs. increased payload capability, increased spacecraft performance, and transportation cost increase of $2.5M).

The fourth option appears most cost effective, for larger instrument payloads, since the preferred mission altitude is maintained, the spacecraft can still be retrieved and the spacecraft cost per pound is minimum at $4.3K/lb.

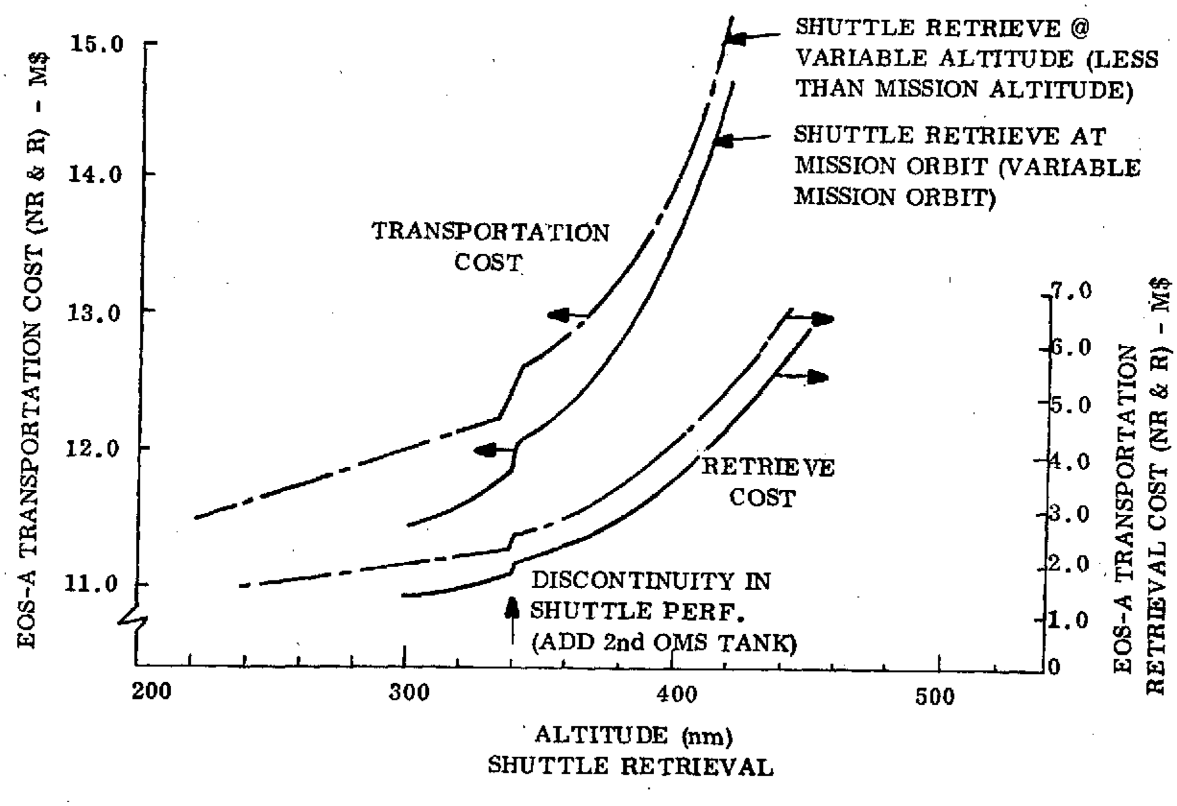

Figure 5-8. EOS-A Transportation Costs

Option #2 which uses the Delta 2910 is still the preferred approach for EOS-A due to its low total transportation cost. System flexibility is maintained to either eliminate retrieval or substitute Delta 3910 if the payload weights exceed the Delta 2910 capability.

Sample transportation cost trades have also been generated for the case with a Shuttle launch and a Shuttle retrieval. These trades establish the relative costs of direct Shuttle launch and retrieval at mission altitude vs. Shuttle delivery and retrieval at a lower altitude using an integral tug to transfer the spacecraft to a higher mission altitude. The two options investigated were direct delivery and retrieval at 360 nm with a total transportation cost of $8.3M and a Shuttle delivery and retrieval at 250 nm with an integral tug transferring the spacecraft to and from the mission altitude of 420 nm with a total transportation cost of $6.1M. This data is summarized in Table 5.10. It becomes evident that transportation costs can be minimized and mission flexibility enhanced by the addition of an integral tug propulsion system to the spacecraft.

Table 5-8. Cost Effectiveness Summary of Alternate Launch Vehicles, Orbit Altitudes & Shuttle Retrieval Altitudes

OPTION #	Launch Vehicle	Mission Altitude	Shuttle Retrieval Altitude	Total Transp. Cost	Allowable S/C wt	Cost /# S/C wt (k$/#)	Allowable Instrument P/L wt ***
1	Delta 2910	360	360	12.4	2590	4.8	880
1a	Delta 2910	360	330	12.2	2550	4.8	840
2	Delta 2910	420	330	12.1	2360	5.1	650
3	Delta 2910	420	420	14.7	2490	5.9	780
4	Delta 3910	420	250	13.6	3175	4.3	1160*
5	Titan IIIB Int. tug	420	250	19.5	4180	4.7	1946*,**

* assumes increased capability spacecraft subsystems and mission peculiar equipment

** provides instrument payload weight capability well in excess of payloads presently under investigation for EOS

*** a weight contingency of at least 160 lbs has been added to spacecraft & mission peruliar weights

Table 5-10. Sample Transportation Cost Trades for Shuttle Launch & Recovery

OPTION #1 Direct Launch & Recovery by Shuttle @ 360 nm

Assume: EOS wt @delivery = 3500 lbs
EOS wt @retrieval = 3450 lbs
Shuttle support wt = 2000 lbs

Delivery Cost

	R	NR
Shuttle Delivery capability = 12,400 lbs		
Cost = $\$4.9M \left(\frac{5550}{12,400}\right)\left(\frac{1}{.78}\right)$	2.8	---
Propulsion System	0.5	1.6

Retrieval Cost

	R	NR
Shuttle Retrieval capability = 10,200 lbs		
Cost = $\$4.9M \left(\frac{5450}{10,200}\right)\left(\frac{1}{.78}\right)$	3.4	---
Total Trans. Cost =	6.7 +	1.6 = $8.3M

OPTION #2 Shuttle Launch to 250 nm - Integral Tug Delivery to 420 nm & return to Shuttle @ 250 nm.

Assume: EOS wt @delivery = 4200 lbs
EOS wt @retrieval = 3450 lbs
Shuttle support wt = 2000 lbs

Delivery Cost

	R	NR
Shuttle Delivery capability = 24,000 lbs		
Cost = $\$4.9M \left(\frac{6200}{24,000}\right)\left(\frac{1}{.78}\right)$	1.6	---
Propulsion System	0.7	2.2

Retrieval Cost

	R	NR
Shuttle Retrieval capability = 21,000 lbs		
Cost = $\$4.9M \left(\frac{5450}{21,000}\right)\left(\frac{1}{.78}\right)$	1.6	---
Total Trans. Cost =	3.9	+ 2.2 = $6.1M

SECTION 6.0

ORBITS/LAUNCH SYSTEM SELECTION

6.1 INTRODUCTION

The selection of the preferred orbit and launch system for EOS-A involves complex trade studies between the type of propulsion system, launch vehicle and orbit altitude and also impacts a wide range of variables. The logic used to simplify this tradeoff was to first make preliminary selections of the type of propulsion system and the launch vehicle. Then, with these preliminary selections already made, the impacts of alternate mission altitudes were evaluated to determine the preferred altitude. It was then necessary to evaluate if any alternate selection of propulsion system or launch vehicle would impact the selection of mission altitude for EOS-A. Finally, the impacts of missions beyond EOS-A were evaluated to determine if these later missions are compatible with the EOS-A selected propulsion system, launch vehicle and mission orbit.

6.2 PROPULSION SYSTEM SELECTION

The selection of the preferred propulsion system for the EOS-A program is discussed in detail in Report #3 "Design/Cost Trades" and summarized in Section 5.0 "Launch System Parametric Analysis" (see Table 5-1 for summary). The selection of an all Hydrazine system has been made for the following reasons:

- It is the lowest cost system investigated.
- It provides maximum system flexibility (compatible with Delta, Titan, or Shuttle launches and a variety of mission orbits and Shuttle retrieval orbits).
- It provides ample growth potential for use on the entire EOS mission model.
- It is a reliable and simple system.
- It minimizes spacecraft design impacts.

6.3 LAUNCH VEHICLE SELECTION

The launch vehicle selection involves a choice between Delta and Titan IIIB NUS since Titan IIID NUS has already been eliminated due to its high cost (see Section 5.0). The allowable spacecraft weights for the three remaining candidate launch vehicles are shown

C-2

in Figure 6-1. The weights shown are the allowable spacecraft weights minus propulsion system weights to provide a common means of comparing the alternate launch vehicles (the Titan requires considerably more weight in the spacecraft propulsion system because it has no upper stage). Two curves are shown for each launch vehicle, the upper curve is the allowable spacecraft weight if the spacecraft is not returned to Shuttle and the lower curve assumes that the spacecraft is returned to Shuttle at an altitude of 300 nm.

Figures 6-2, 6-3 and 6-4 present the performance of Delta 2910, 3910 and Titan IIIB showing their compatibility with Shuttle retrieval and alternate weight and capability spacecraft. Figure 6-2 shows that the Delta 2910 launch vehicle is compatible with direct Shuttle retrieval to altitudes up to 440 nm but requires a light weight spacecraft (as defined in Section 2.0 Table 2.8) if a mission orbit altitude of 420 is selected. The Delta 3910 launch vehicle is still compatible with direct shuttle retrieval at mission altitudes exceeding 420 nm while maintaining considerable weight margin for the nominal capability EOS-A as defined in Table 2.8. The TitanIIIB shown in Figure 6-4 is compatible with the maximum capability EOS-A defined in Table 2.9 of Section 2.0 and includes resupply provisions and two wideband tape recorders although direct Shuttle retrieval at a mission altitude of 420 nm becomes marginal.

Since there is more than a 4M dollar cost differential between a Delta launch and a Titan IIIB launch and a reasonable EOS-A spacecraft can be launched on Delta at the preferred mission altitude of 418 nm, Delta has been selected as the preferred launch vehicle for EOS-A. A layout of an EOS-A Delta spacecraft is presented in Section 2.0, Figure 2-3. The selection between Delta 2910 and 3910 can be made when the instrument designs and weights have been better defined, the final mission orbit has been selected, and the need to retrieve the spacecraft via Shuttle has been resolved.

6.4 MISSION ALTITUDE SELECTION

Once the spacecraft propulsion system has been selected (hydrazine) and the launch vehicle has been selected (Delta) the impacts of alternate mission altitudes can be evaluated using the evaluation criteria presented in Section 2.6 and summarized in Table 6-1. This table

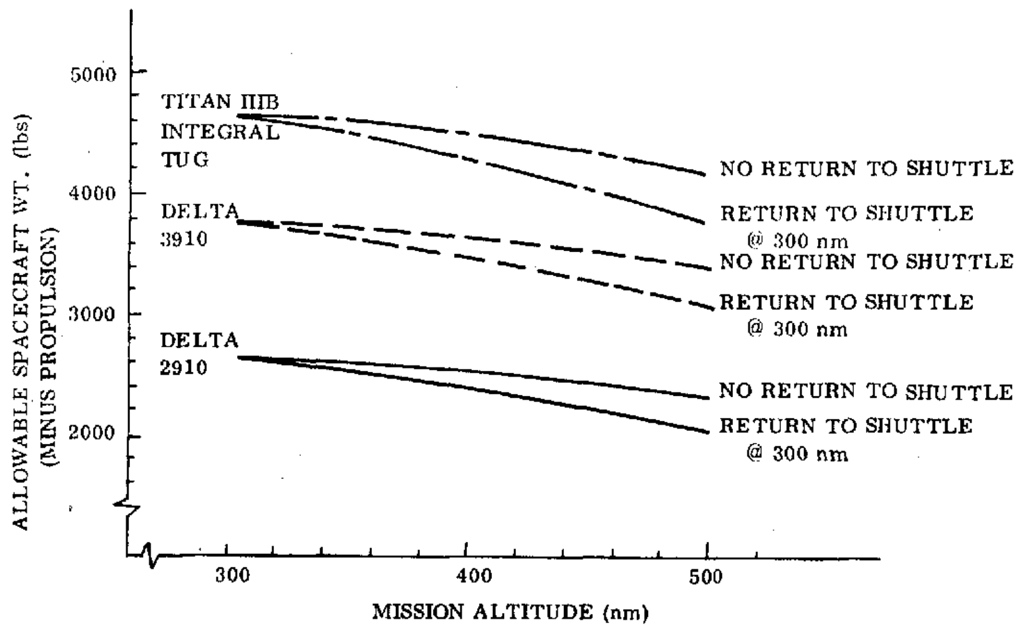

Figure 6-1. EOS-A Launch Vehicle Performance
(Hydrazine Propulsion System)

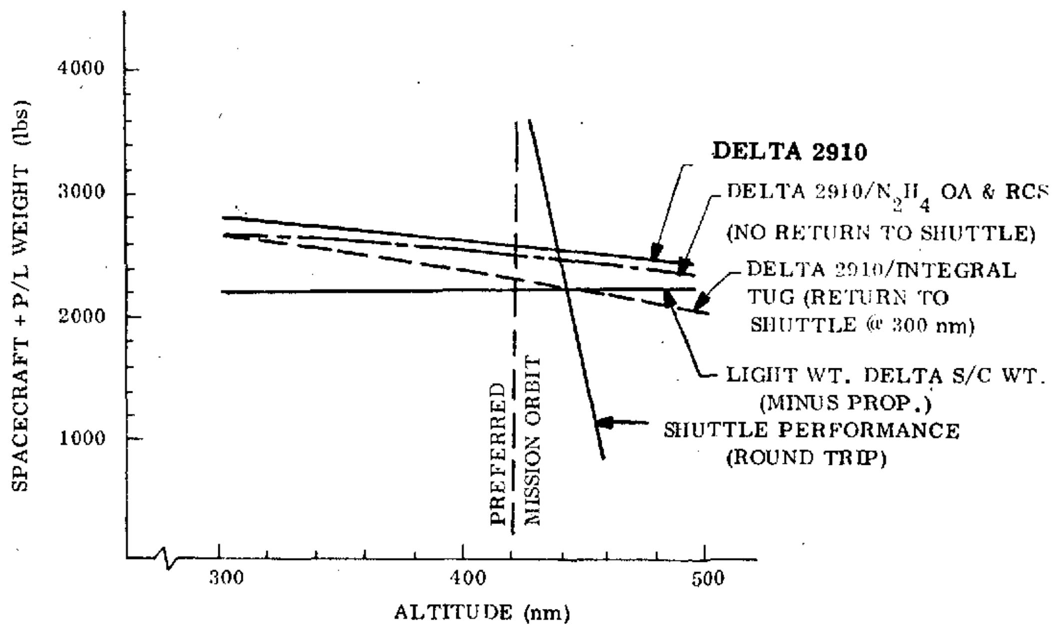

Figure 6-2. Delta 2910 & Delta 2910-Integral Tug Performance

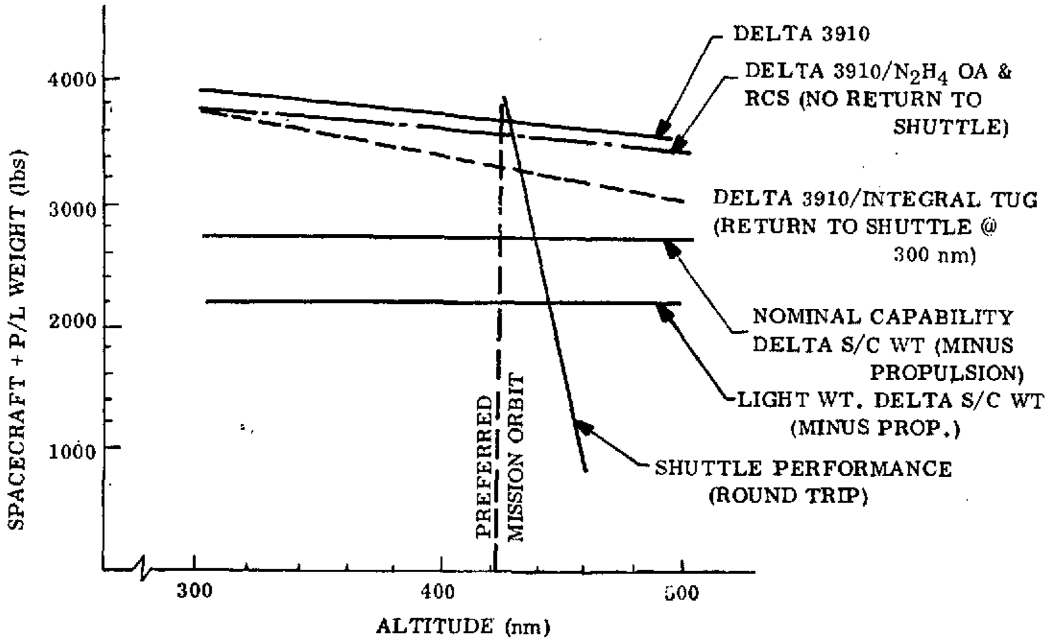

Figure 6-3. Delta 3910 & Delta 3910-Integral Tug Performance

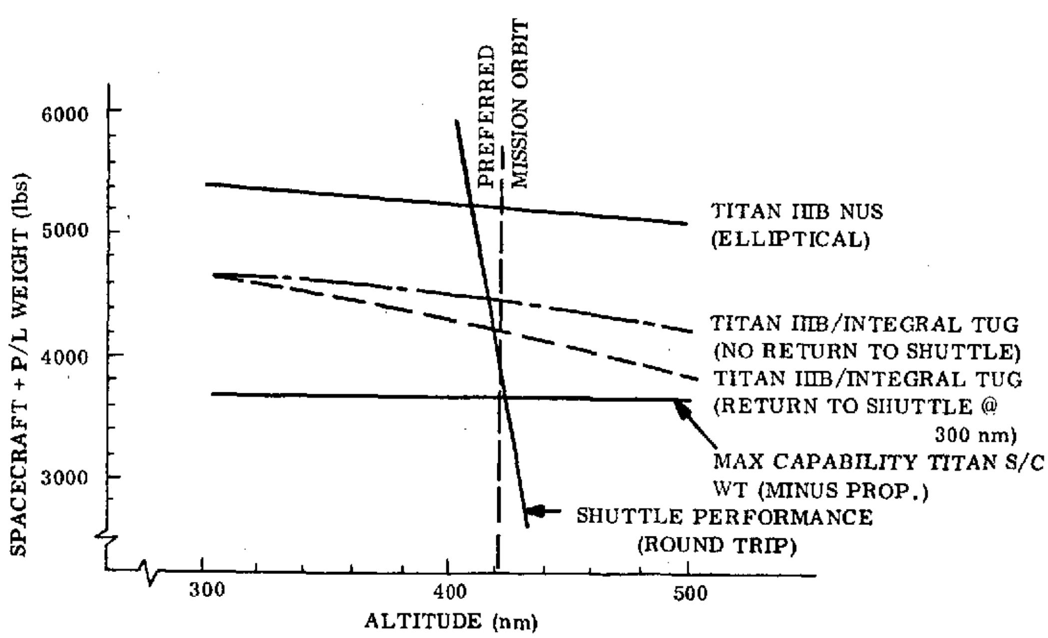

Figure 6-4. Titan IIIB NUS & Titan IIIB-Integral Tug Performance

Table 6-1. Tradeoff Summary for EOS-A Orbit Selection

(Delta Launch, Hydrazine Propulsion)

Evaluation Criteria	Req Or Range	Candidate Orbits 359	386	399	418	425
o Shuttle Compatibility		Good	Good	Good	Good	Fair
Direct Shuttle Access	3700#	8500#	6300#	5100#	3400#	2800#
Use of Prop. System@330 nm	3700#	13200#	13200#	13200#	13200#	13200#
Wt Penalty to Return to 330 nm	----	40#	70#	100#	140#	150#
o Mission Compatibility		Good	Fair	Fair	Good	Good
Global Coverage	----	Yes	Yes	Yes	Yes	Yes
Realtime Coverage (USA)	----	Yes	Yes	Yes	Yes	Yes
Minimum Sidelap	5-15%	8.1%	12.5%	12%	11.5%	5.7%
Repeat Cycle	15-18 Days	16 Days	17 Days	17 Days	17 Days	16 Days
Access Time	2-4 Days	3 Days	2 Days	2 Days	3 Days	3 Days
Minimum Offset Pointing	$\leq 30^{\circ}$	37°	45°	45°	32°	32°
o Ground Station Compatibility		Poor	Fair	Good	Good	Good
No. of Grd Station Req'd	3	4	4	3	3	3
Freq of Orbit Adjusts	$<$1/Two Cyc.	$>$1/Two Cyc.	$>$1/Two Cyc.	$>$1/Two Cyc.	$<$1/Two Cyc.	$<$1/Two Cyc.
o Launch System Impacts		Good	Good	Good	Fair	Fair
Transp Costs (Ret@Miss. Orb.)	----	12.4M	13.4M	13.6M	14.7M	15.0M
Transp Costs (Ret@330 nm)	----	12.1M	12.1M	12.1M	12.1M	12.1M
System Felxibility	----	Minor Impact By Orbit Selection With Range 359 to 425				
Ease of Transition to Shuttle	----	All Launches Compatible With Shuttle Launch & Retrieve@250 nm				
Launch System Perf. (All. S/S Wt) (Assume Retrieve@ 330 nm)	2215#	2515#	2485#	2430#	2355#	2330#
o Spacecraft Impacts	Minor Impacts On S/C Cost Or Weight As A Function of Orbit Altitude In Range 359 To 425 nm.					
o Payload Instrum. Impacts	Minor Impacts On Instrument Cost Or Weight As A Function Of Orbit Altitude In Range 359 To 425 nm When Engineering Judgement Used.					
o Impacts of Later Missions	Minor Impacts From Later Missions On Selection Of EOS-A Orbit (Delta L/V & Hydrazine Propulsion Compatible With Later Missions)					

Summary: 418 nm Orbit Selected As Best Meeting Mission & Ground System Compatibility While Having Acceptable Launch System Impacts indicates the selection of 418 nm as the preferred EOS-A mission altitude. The rational for this selection follows discussing the impact of each of the criteria used for the evaluation. The five alternate mission altitudes selected from the mission analysis (Section 3.0) are 359, 386, 399, 418 and 425 nm.

6.4.1 SHUTTLE COMPATIBILITY

When an on-board hydrazine propulsion system with orbit transfer capability is added to the spacecraft all orbits under consideration are compatible with Shuttle retrieval at 330 nm which has been determined as the cost effective retrieval altitude for spacecraft launched with limited capability expendable launch vehicles. Direct Shuttle access for spacecraft weighing in excess of 3700 lbs are available for the three lowest candidate orbits while Shuttle can deliver and retrieve a spacecraft weighing 3400 lbs to 418 nm and retrieve a spacecraft weighing in excess of 3700 lbs. Considering all factors involved the four lower orbits under consideration, 359, 386, 399 and 418, have good Shuttle access compatibility with the 425 orbit only slightly poorer.

6.4.2 MISSION COMPATIBILITY

The candidate orbits, 359, 418 and 425 nm, have acceptable compatibility with the mission constraints while the 386 and 399 orbits are less acceptable. The 359 nm orbit meets all compatibility requirements with the exception of exceeding the desired minimum offset pointing for HRPI. The 418 orbit slightly exceeds the desired limits of minimum sidelap (by 1.5%) and minimum offset pointing (by 2^{o}) but is still considered a good fit to the mission compatibility. The 425 orbit is possibly the best from a mission viewpoint since it only exceeds the desired maximum HRPI offset pointing angle (also by 2^{o}) while meeting the remainder of the mission constraints.

The two remaining orbits 386 and 399, are considered less acceptable due to their larger sidelaps 12.5 and 12 respectively, which increases the amount of ground processing but particularily because the large offset pointing angle (45^{o}) required for HRPI.

6.4.3 GROUND SYSTEM COMPATIBILITY

The ground system compatibility is directly proportional to orbit altitude with: 1) the higher orbits preferred due to minimum drag and therefore less frequent orbit adjusts and 2) the reduced number of ground stations required for real time coverage of the USA. The present ERTS system can be used only as low as 490 nm at which point real time coverage no longer exists in the south central portion of the USA. As the altitude decreases, the amount of real time coverage missed in this area increases. By substituting an alternate ground station (Merritt Island in the STDN network or even Sioux Falls) for NTTF the real time coverage of the USA can be achieved with three stations down to an altitude of approximately 400 nm. Below 400 nm a fourth station is required for real time coverage at a penalty of approximately 0.5 million dollars to add wideband capability. Therefore, the orbit altitude preference for ground station compatibility is in inverse order to their altitude, 425, 418, 399, 386 and 359.

6.4.4 LAUNCH SYSTEM IMPACTS

The orbits preferred for launch system impacts are again related to orbit altitude but with the lower altitudes preferred. By using an on-board propulsion system, the transportation costs can be made independent of selected orbit altitude at approximately $12M per launch and retrieval. These costs can be reduced by $.4M if retrieval is not required. The transportation cost analysis is discussed in detail in Section 5.9. The major impact in the higher altitudes is the limiting allowable spacecraft weight which still exceeds the requirement for a light weight Delta 2910 launch by over 130 lbs. This weight margin can be increased to almost 300 lbs if Shuttle retrieval is eliminated for EOS-A.

6.4.5 SPACECRAFT IMPACTS

The impacts on spacecraft subsystem costs as a function of altitude are discussed in detail in Section 4.0. It was determined that there was very little impact on spacecraft subsystem recurring costs or weight for the ranges of orbit altitudes under consideration for EOS-A. This, of course, is the desired result for a flexible spacecraft design.

6.4.6 PAYLOAD INSTRUMENT IMPACTS

The cost impacts on payload instruments are discussed in detail in section 4.0. In practice instrument designs are expected to be based on a few descrete aperture sizes. Performance will be allowed to vary over small altitude ranges causing stepped cost curves. The instrument costs are therefore expected to remain relatively constant over the small range of altitudes under consideration for EOSA.

6.4.7 IMPACTS OF LATER MISSIONS

The later missions defined in Table 6-2 indicate that the selection of Delta as the launch vehicle for EOS-A and a mission orbit altitude of 418 is compatible with the mission model recommended in Table 6-3. This mission model indicates that all proposed missions can be accommodated with a Delta or Shuttle launch and that later missions using a Thematic Mapper (EOS-C) can also use the 418 nm orbit selected for EOS-A.

Table 6-2. Traffic Model

	77		78				79				80				81				82			
	3Q	4Q	1Q	2Q	3Q	4Q	1Q	2Q	3Q	4Q	1Q	2Q	3Q	4Q	1Q	2Q	3Q	4Q	1Q	2Q	3Q	4Q
OS-A							▽ Launch															
OS-B												▽ Launch										
OS-C																▽ Launch						
OS																	▽ Launch					
LAR MAX.				▽ Launch																		
ASAT	▽ A Launch								▽ B Launch													
ERS															▽ 2 Spacecraft					▽		

Table 6-3. Mission Module

Mission	Orbit Alt Inc (nm) (deg)	Launch Date(s)	Space-craft wt (lb)	Launch Vehicle Capability (lb)	Launch Vehicle	Comments
EOS-A	418 (SS)	1979	2380 ①	2580 ③	Delta 2910	Retrieval with Shuttle
EOS-B	450 (SS)	1980	2439 ②	2500 ③	Delta 2910	" " "
EOS-C	418 (SS)	1981	2512 ①	2580 ③	Delta 2910	" " "
Shuttle Demo	300 (28.5)	1980	<4000	51,000	Shuttle	Resupply Demo & Retrieve
SEOS	GEOSYNC (2^o)	1981	2716	3600 OOS 7200 TUG	Shuttle/Tug	Retrieve or Replace with Tug
SOLAR MAX	285 (30^o)	1978	3450	3900	Delta 2910	Retrieve with Shuttle
SEASAT A	430 (108^o)	1977	2050	2420	Delta 2910	No Retrieve
SEASAT B	324 (90^o)	1982	2230	2800	Delta 2910	No Retrieve
5 Band MSS	500 (SS)	1978	2250	2400	Delta 2910	No Retrieve
EOS-A Ⓡ	418 (SS)	1982, (etc)	Not Defined	>4000	Shuttle	Retrieval or Resupply Decision Open
EOS-B Ⓡ	450 (SS)	1983, (etc)		>4000	Shuttle	
EOS-C Ⓡ	418 (SS)	1984, (etc)		>4000	Shuttle	

① No WB Tape Recorders
② Reduced Instrument Complement
③ Significant Increase if use Delta 3910